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(In Two Parts)

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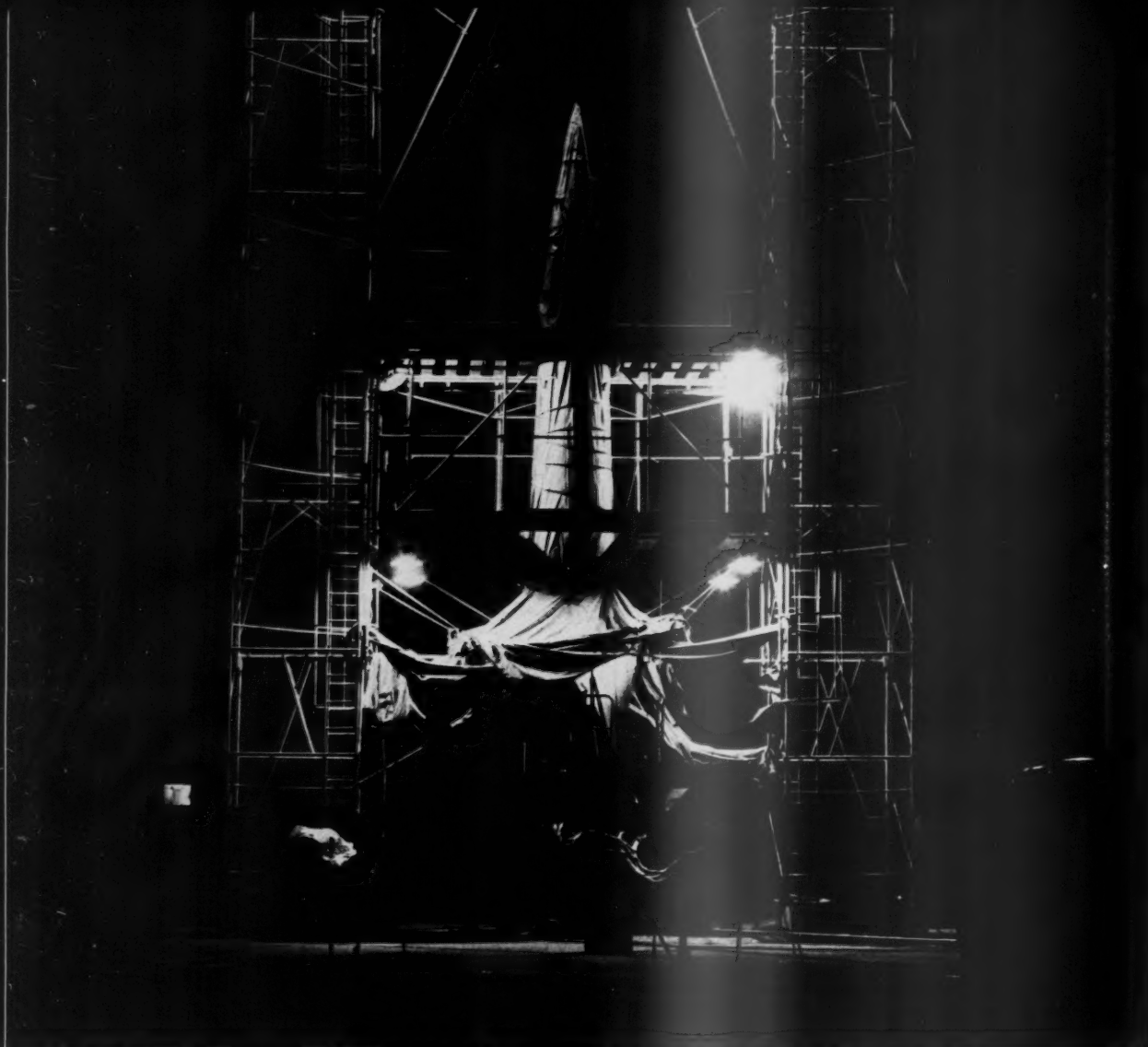
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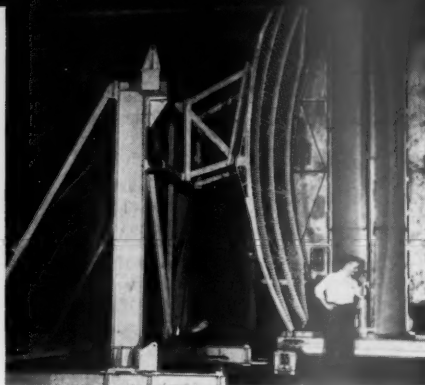


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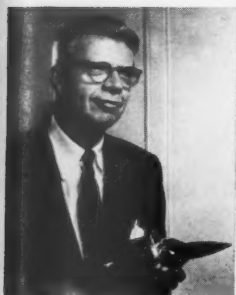


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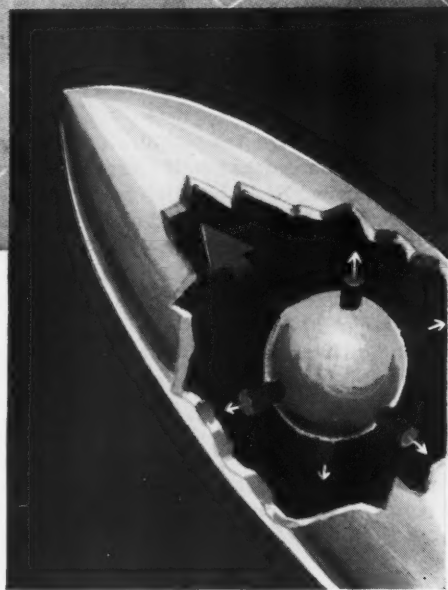
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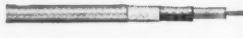
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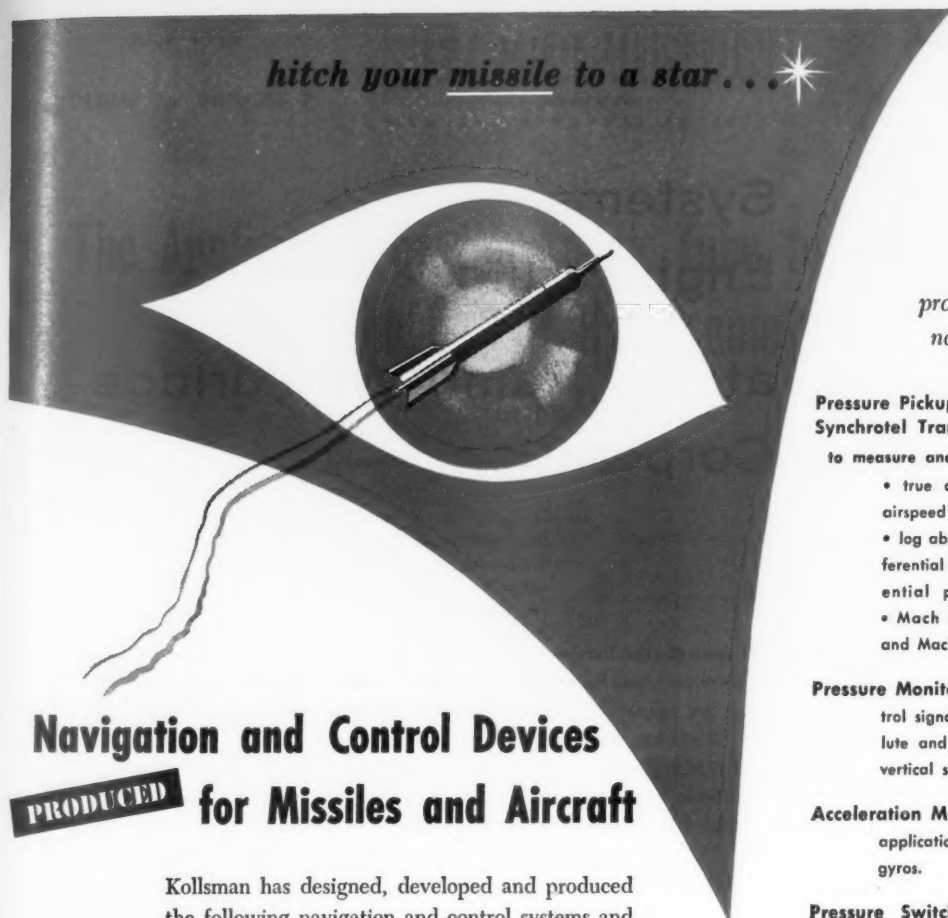
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The Application of the Shock Tube to the Study of the Problems of Hypersonic Flight¹

A. HERTZBERG²

Cornell Aeronautical Laboratory, Inc., Buffalo, N. Y.

This paper discusses some of the new scientific and technical problems introduced by the high temperature conditions encountered in hypersonic flight. Modifications of the shock tube which have been studied at Cornell Aeronautical Laboratory for the investigation of these problems are presented. The shock tube is one of the few laboratory instruments capable of generating the high temperature conditions in air encountered in hypersonic flight studied. The conventional shock tube is limited since the maximum flow Mach number that can be achieved behind a normal shock wave in air is approximately 3. When the conventional shock tube is terminated by an expansion nozzle, high Mach number flows can be achieved as well as high stagnation temperatures. Actual flow conditions can therefore be closely simulated for the investigation of heat transfer rates and other aerodynamic problems. The application of the shock tube to other areas of high temperature research is briefly discussed. In particular, a technique for the study of high temperature chemical reaction rates is described.

Introduction

FLIGHT within the earth's atmosphere at Mach numbers of 10 and above introduces a number of new and interesting scientific and technical problems not associated with flight at lower Mach numbers. The most important of these problems arise from the extremely high temperatures which are encountered in the field of flow around a body resulting from the conversion of the kinetic energy associated with the flight velocity into thermal energy across the shock wave. At the high temperatures the thermal energy or enthalpy of the air is sufficient to excite the higher vibrational states of the diatomic molecules, produce dissociation of the nitrogen and oxygen molecules, cause chemical reactions such as the formation of nitric oxide, and induce thermal ionization of some of the molecular and atomic species of which the air is composed.

The investigation of hypersonic flight problems is complicated by the changes in gas composition which occur at these high temperatures. Fig. 1 shows the maximum temperature encountered at the forward stagnation point on a blunt

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¹ The work to be described was supported by the Air Force through the Arnold Engineering Development Center; the Aeronautical Research Laboratory, Wright Air Development Center; and the Office of Scientific Research; and represents the contribution of a large number of the personnel in the Aerodynamic Research Department of the Cornell Aeronautical Laboratory.

² Head, Gasdynamics Section, Aerodynamic Research Department.

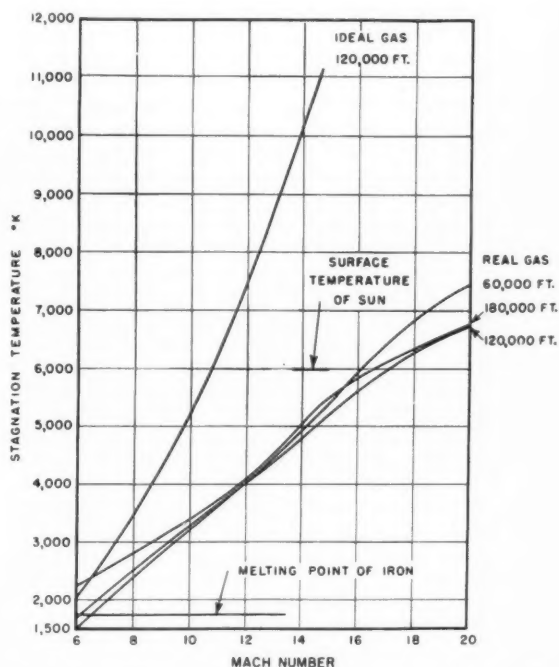


Fig. 1 Maximum temperature at the blunt nose of a perfectly insulated body

nosed, perfectly insulated body at hypersonic velocities. As can be seen in this figure, at a flight Mach number of 16 the stagnation temperature is equivalent to the surface temperature of the sun. The large differences between the temperature calculated assuming an ideal gas with a constant specific heat ratio of 1.4 and the equilibrium temperature calculated using the actual composition can also be observed. The mole fraction equilibrium composition of air at elevated temperatures and at sea level density is shown in Fig. 2. The calculations below 4000 K are the early work of Hirschfelder and Curtiss (1)³ using the then accepted value for the dissociation energy of nitrogen, 7.37 electron volts per molecule. The calculations above 4000 K by the National Bureau of Standards (2) are carried out using the recently determined value, 9.76 electron volts per molecule (3). The effect of the change in the dissociation energy of nitrogen is evidenced by the change in fraction of atomic nitrogen at 4000 K. The

³ Numbers in parentheses indicate References at end of paper.

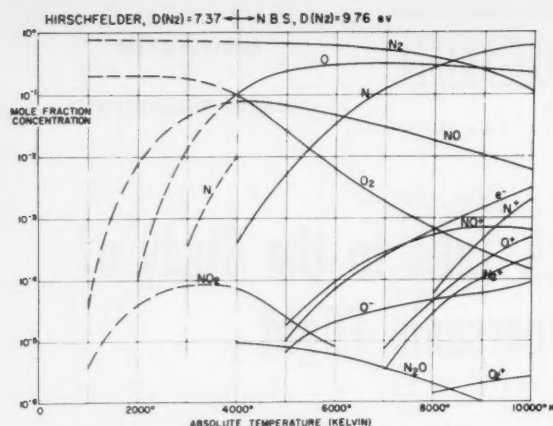


Fig. 2 Mole fraction equilibrium composition of air at elevated temperatures and at sea level density

use of the wrong dissociation energy appreciably affects gas dynamical calculations (18). At 6000 K it may be observed that, in addition to nitrogen and oxygen molecules, the air consists of atomic oxygen, atomic nitrogen, small percentages of the various oxides of nitrogen, electrons, negative oxygen ions, and positive ions of nitric oxide. The major portion of the electrons below about 8000 K arises from the thermal ionization of the small but important concentration of NO. The small quantity of some of the constituents such as electrons, although not of such magnitude to radically alter the thermodynamic properties, may be sufficient to appreciably alter the transport phenomena (heat conduction, viscosity, etc.) at these high temperatures. For example, at a temperature of 4000 K which corresponds to a flight Mach number of 12, the electron concentration behind a normal shock wave is sufficient to produce an electrical conductivity equivalent to that of sea water. It is to be expected that the presence of the free atoms, electrons, and molecules in excited states will also influence flow phenomena such as boundary layer formation as a result of the energy transported by these species existing at the high temperature. The changes in the composition of air at the elevated temperatures of hypersonic flight directly affect not only stagnation temperature, but other aspects of body aerodynamics (4). This is strikingly illustrated by comparison of real and ideal limiting wedge angles for shock wave attachment (Fig. 3).

In the above discussion an equilibrium state was assumed in all calculations. However, this assumption in some of the problems associated with hypersonic flight may be incorrect since the transit time of a particle of gas near the body may be of the same order as the relaxation time. Lack of knowledge of high temperature reaction rate phenomena, molecular, atomic, electronic, and ionic scattering cross-section data or interaction potential data at the present time considerably complicates the problem of determining the nonequilibrium properties of air. Indeed, it is this phase of the problem which vitally requires basic research.

In view of the chemical and physical activity of air encountered in hypersonic flight, it is important that laboratory studies of flow at very high Mach numbers be done in air at the appropriately high temperatures.

In a conventional wind tunnel the stagnation temperature at the test section is approximately equal to the plenum temperature. Therefore heating techniques must be developed which yield flight stagnation temperatures. The modified pebble heater (5) is perhaps the most successful of the direct heating methods. Temperatures of 1800 K, corresponding to a free-flight Mach number of approximately 6, have been achieved. It is also possible to heat air to relatively high stagnation temperatures using various types of electric arcs

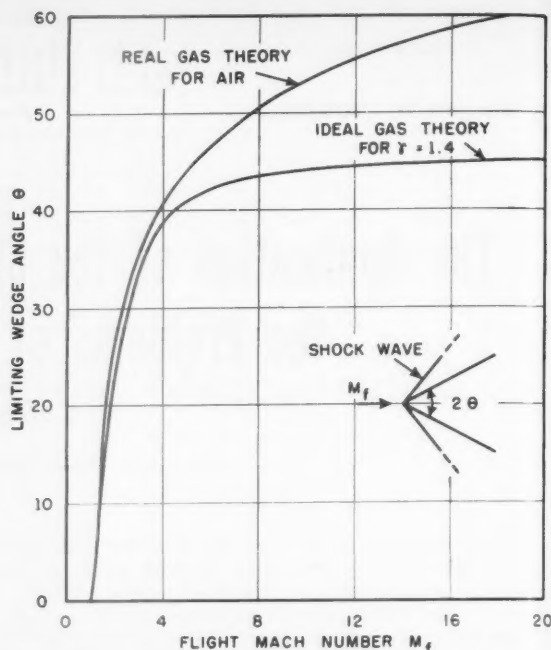


Fig. 3 Limiting wedge angle for shock attachment in air. Curve for air is for ambient pressure of 0.01 atmospheres and ambient temperature of 298 K

(6), and temperatures of about 3600 K have been reported in arcs used to produce nitric oxide from air.

The stagnation temperature in conventional steady flow wind tunnels, however, is limited by the strength at high temperatures of present-day materials. Temperatures required for the simulation of Mach 7 flight are well above the melting points of all known metals. Considerable effort is therefore being devoted to the development of other experimental techniques which circumvent these limitations.

Nonsteady heating methods, such as rapid compression by means of a piston, offer means of reaching high temperatures for brief periods which would not damage the tunnel structure. Another technique for rapid heating, which has in recent years found increasing application, involves the use of strong shock waves; temperatures beyond those encountered in flight may be generated with relatively simple shock tube equipment (7). One of the chief attributes of the shock tube as a generator of hot gases is the relative precision with which the state of the test gas is known, assuming that the thermodynamic equilibrium is attained and that the shock Mach number can be adequately measured. This paper describes the use of a shock tube for the study of high stagnation temperature flows and high Mach number flows in air. In addition, the application of shock tube techniques at Cornell Aeronautical Laboratory to other areas of high temperature research is outlined.

The Conventional Shock Tube

Before describing the specific modifications of the shock tube required to generate hypersonic flow, it may be appropriate to review briefly the phenomena that occur in a conventional shock tube and describe its potentialities as a laboratory instrument for the investigation of high temperature problems. The conventional shock tube consists of a constant-area duct separated by a thin diaphragm into regions of high and low pressure. When the diaphragm is ruptured, a shock wave is generated which propagates through the low pressure gas, accelerating the gas and simultaneously raising its temperature. If the shock wave propagates

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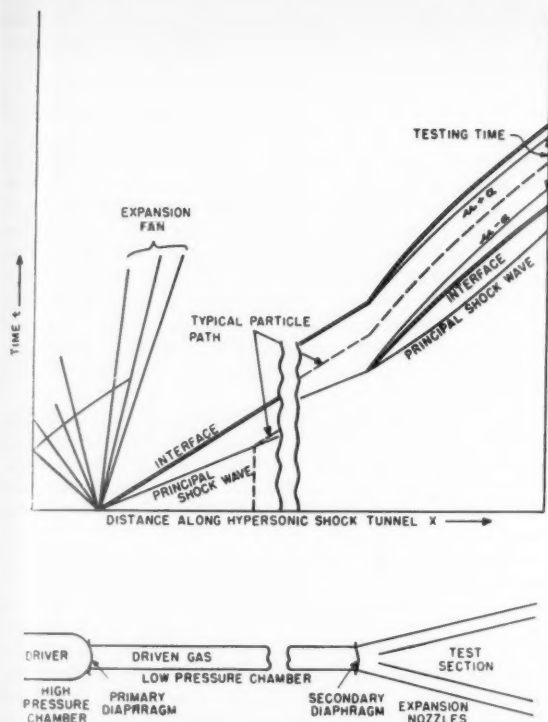


Fig. 4 Wave diagram for a shock tube nozzle configuration

through the tube without attenuation due to viscous losses or other causes, then the gas between the interface (which ideally is a plane separating the gas of the driver section and the gas in the low pressure section) and the shock is then at a uniform temperature and velocity. At any point in the shock tube the flow useful for testing is that between the shock wave and the interface. The essential features of the flow in the shock tube can be observed in the wave diagram of Fig. 4.

Although supersonic flow Mach numbers can be achieved in a straight shock tube behind the normal shock wave, the flow Mach number attainable behind a very strong shock wave is limited. Although an extremely high velocity can be generated behind a normal shock in a constant area duct by using a large initial pressure ratio, the ambient temperature behind the shock is correspondingly high. This consideration yields a maximum Mach number of 1.89 for an ideal diatomic gas (8). Fig. 5 shows the observed variation with shock Mach number of the flow Mach number behind a normal shock moving into still air. The points shown in the diagram are measured flow Mach number values in a straight shock tube. Since the excitation of the higher vibrational states of the molecules and dissociation absorb considerable amounts of the available energy, the static temperature behind the normal shock does not rise so rapidly as if the gas were ideal. Consequently, the flow Mach number behind the normal shock in air is higher.

Fig. 6 is a schlieren photograph of flow phenomena in a straight shock tube with the normal shock wave moving from left to right into still air. For the measured shock Mach number of 9 in this figure, the temperature behind the shock wave is about 4000 K. The flow Mach number may be determined from the measurements of the Mach angle of the disturbance waves which can be seen behind the shock. The shaded region behind the normal shock is due to a relaxation time to equilibrium flow of approximately 10 microsec. The straight shock tube studies described are a part of the investigations being conducted at Cornell Aeronautical Laboratory of the equilibrium and nonequilibrium phe-

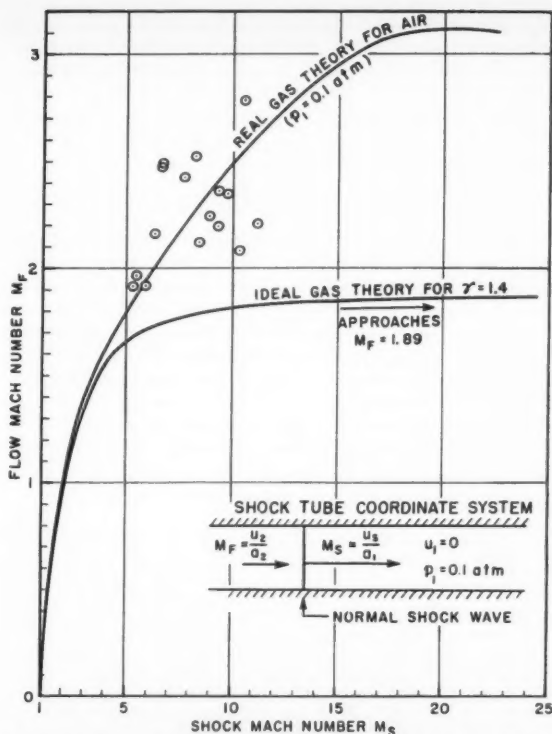


Fig. 5 Flow Mach number obtainable in a straight shock tube



Fig. 6 Schlieren photograph of air in a straight shock tube; vertical knife edge

nomena behind strong normal shock waves.

The most important limitation of the shock tube aside from the Mach number limitation is the extremely brief period during which high temperatures can be sustained. The conventional shock tube is, nonetheless, a valuable tool for investigation of high temperature phenomena in air when it is not necessary to simulate the flow Mach number. It provides high density and high Reynolds number conditions.

In order to generate strong shock waves, the practice at Cornell Aeronautical Laboratory has been to generate the initial high pressures required behind the diaphragm by combustion of gaseous mixtures of oxygen, hydrogen, and a diluent. In early investigations, diaphragm strengths were chosen so that the diaphragm would be ruptured prior to achievement of constant volume combustion pressures (9), with the expectation that the final stages of combustion would take place at constant pressure. While strong shock waves in high density gas were achieved by this "constant pressure" drive, measurements of the shock velocity and pressure in the flow behind the shock wave indicated that irregularities in the combustion process could result in appreciable shock wave attenuation. Expansion waves generated in the heating zone during combustion could overtake and weaken the normal shock. The most serious con-

sequence of this attenuation is the resultant change in the flow parameters behind the shock. Under these circumstances a steady flow condition is not achieved in the shock tube. Nevertheless, from the shock velocity record or the measure pressure history of the flow, the actual variation of flow conditions with time can be calculated by means of the techniques of the method of characteristics (10).

The Hypersonic Shock Tunnel

The chief limitation of aerodynamic testing in the conventional shock tube is the flow Mach number limitation previously discussed. Higher flow Mach numbers can be achieved by modifications of a conventional shock tube (8). For example, if a conventional shock tube is terminated by a supersonic nozzle (11) as indicated schematically in Fig. 4, the steady supersonic flow behind the shock wave expands through this nozzle and the flow Mach number can thus be raised to the desired test Mach number. By controlling the strength of the primary shock and employing the proper nozzle configuration, the stagnation temperature and Mach number conditions of free flight can be duplicated.

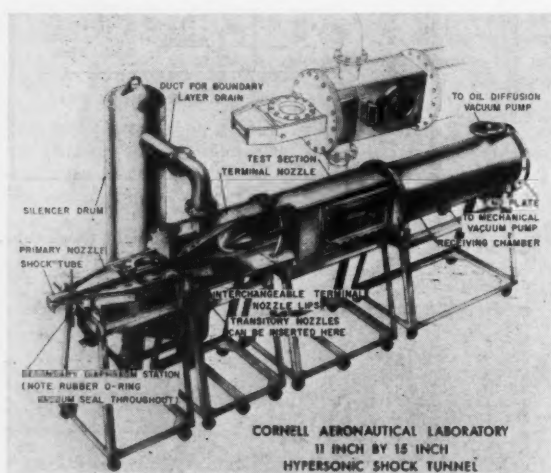


Fig. 7 Cornell Aeronautical Laboratory 11- x 15-in. hypersonic shock tunnel

Fig. 7 is a drawing of the 11- x 15-in. hypersonic tunnel at Cornell Aeronautical Laboratory which is being employed for hypersonic testing at the present time. The flow expansion occurs through a two-step nozzle and the boundary layer generated along the walls of the shock tube and the primary nozzle is removed at the beginning of the second step. The test section indicated by the location of the cone model is 11 x 15 in. and is equipped with 10-in.-diam observation windows. The rectangular section with the window may also be used as a test station.

In the design of a hypersonic nozzle, real gas calculations must be used since the expansion ratio required to achieve a given test Mach number is appreciably altered (4). The nozzle expansion ratio is greatly increased in comparison to the value required to produce the same flow Mach number assuming an ideal gas with a specific heat ratio of 1.4. The Reynolds number per foot in the test section is correspondingly reduced in comparison to the ideal gas values. As an indication of the extent of these effects, at a flow Mach number of 10, the expansion ratio is increased and the Reynolds number per foot decreased by a factor of 10. The effect on testing time is also of some importance since the flow duration in any reasonable length of tube is of the order of milliseconds. When real gas properties are considered, the testing time is reduced by a factor of 2 at high Mach numbers as compared with the testing time calculated on an ideal gas basis. For a given shock strength in the real gas,

the flow velocity is greater than in the ideal gas and, consequently, all the useful flow passes a given point in the shock tube in a shorter time.

At the end of the low-pressure section it is necessary to insert a secondary diaphragm. This diaphragm is required in order that the nozzle section may be evacuated to enable rapid establishment of the high Mach number flows. When the shock propagates into the nozzle, disturbances are generated which tend to move upstream against the flow, and reduce the available testing time. By properly selecting the initial pressure in the nozzle, only weak disturbances will be generated which will be swept out of the nozzle with the characteristic velocity ($u - a$) where u is the flow velocity and a is the velocity of sound. This enables the maximum use of the available testing time to be achieved in the hypersonic nozzle. A detailed discussion of these flow-starting phenomena is given in (12).

Fig. 8 shows the experimental variation of the tunnel flow Mach number with time during a run as determined by measurements of the shock angle on a 20 deg wedge. These measured Mach numbers are believed to be accurate to ± 0.2 of a Mach number. The initial time indicated here is the time at which the shock wave first crosses the wedge. It may be interesting to note that, prior to the establishment of steady flow, a very high Mach number is obtained in the test flow as predicted by theoretical calculations of the starting phenomena.

When the waves generated in the starting process are swept downstream, a period of steady flow of approximately one millisecond duration is established as indicated in Figs. 8 and 9. Mach number measurements obtained from records, as in Fig. 9, show that, within measurement errors

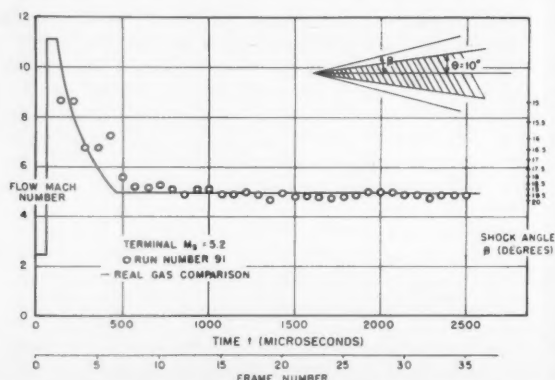


Fig. 8 Development of flow about a wedge in the terminal nozzle of the 11- x 15-in. hypersonic shock tunnel

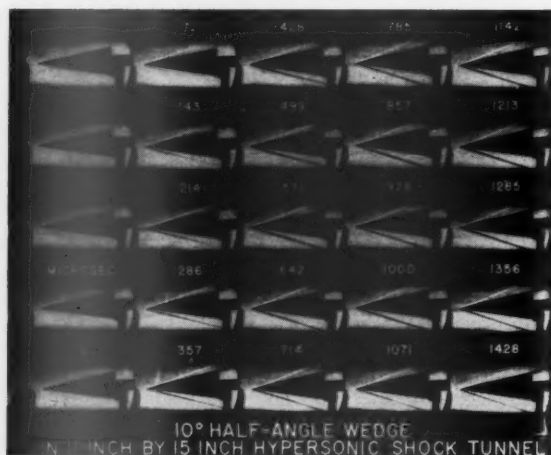


Fig. 9 20° wedge in 11- x 15-in. hypersonic shock tunnel; run 35

of the shock angle, the flow Mach number may be considered constant during the testing time.

Although the flow Mach number has been observed to be relatively constant in the test section, it is possible for the test section density and pressure to vary considerably if large shock attenuation occurs in the constant area low pressure section. As previously described, the expansion waves which cause the shock attenuation also produce a changing condition in the flow behind the shock wave. This flow attenuation appears primarily as a continuous change in pressure and density of the flow entering the nozzle, resulting in a varying pressure and density at the test model. The variation in density and pressure in the test section cannot be observed from measurements of flow Mach number alone.

No satisfactory instrumentation is presently available for the measurement of the low density and pressure established in the test section. This is one of the chief limitations of testing in the hypersonic shock tunnel. However, it has been found possible to calculate the tunnel flow conditions from careful measurements of the shock Mach number and the pressure history of the flow behind the normal shock in the low pressure section and from measurements of the flow Mach number in the test section. The theoretical calculations are carried out using the method of characteristics (10).

At the present time, the driver technique is being modified to permit the attainment of constant volume combustion pressure. Fig. 10 shows the calculated densities which can be produced in the hypersonic shock tunnel using the "constant volume" combustion of a mixture of 81 per cent helium and 19 per cent stoichiometric oxygen and hydrogen. This driver has a greater acoustic velocity than any mixture of stoichiometric oxygen and hydrogen with excess hydrogen.

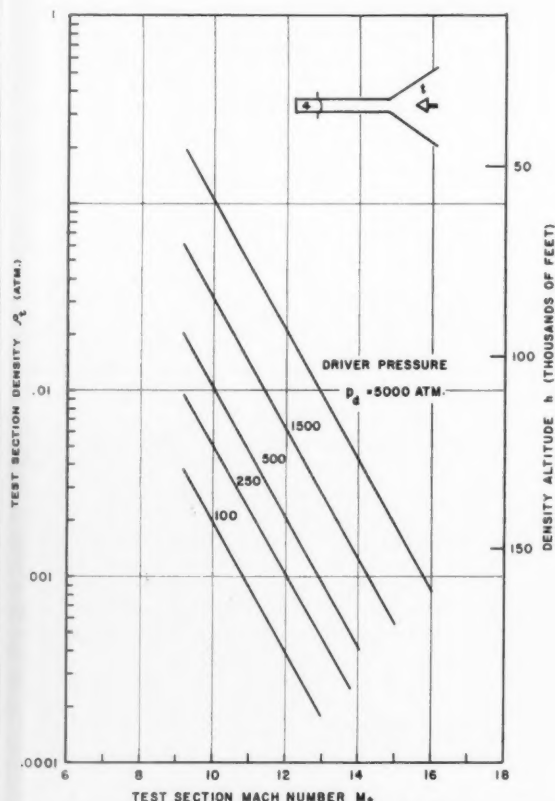


Fig. 10 Density in the test section for several driver pressures established by combustion at constant volume of a mixture of 81% helium and 19% stoichiometric oxygen and hydrogen $\gamma = 1.45$

It is anticipated that the constant volume process will yield less shock attenuation than is observed with the "constant pressure" process.

Methods of Heat Transfer Measurement in the Hypersonic Shock Tunnel

Because of the importance of knowledge of the heat transfer rates in high velocity flight, the first series of measurements that have been carried out in the hypersonic shock tunnel have been measurements of heat transfer rates on models mounted in the nozzle. Since the available testing time in the hypersonic shock tunnel is on the order of one millisecond, and the density is low, conventional heat transfer rate instrumentation cannot be used and special techniques have been developed.

The maximum temperature rise on models fabricated from insulating materials and mounted in a shock tunnel is about 100 K during the brief testing time. By measuring the time history of the surface temperature during this interval, the heat transfer rate at the surface can be determined by calculation.

In shock tunnel applications the temperature rise is measured by a thin metallic film on the order of one micron in thickness bonded to the surface of the model. If this film is one micron thick or less, the temperature of the film will correspond to that of the surface with a lag which may be considered small as compared to the testing time. The temperature rise in the film is measured by observing the change in its electrical resistance. Indeed, the rise time of these gages is so fast that the gages have been successfully used as shock wave detectors for the purpose of measuring shock velocity (13). Gages of this type have also been used to measure heat transfer rates to the walls of the shock tube (14). Basically, the technique employed is a modification of the evaporated film thermocouple which has been developed in recent years for the measurement of transient temperatures in gun barrels (15).

At Cornell Aeronautical Laboratory, a simple and effective thin film gage has been developed by painting a suspension of silver and platinum on glass or quartz test models which are then baked at approximately 1100 F for 20 min (16). Measurements of the thickness of such gages have indicated that they are of a uniform thickness of about one micron. At the present time at Cornell Aeronautical Laboratory, gages

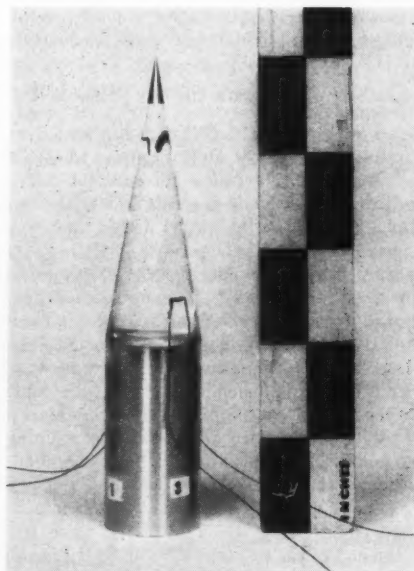


Fig. 11 Resistance thermometer mounted on a 20° fused quartz cone

of this type are being used in a variety of heat transfer programs carried out in the hypersonic shock tunnel.

The performance of these gages prior to their application to the hypersonic range has been verified by tests in a straight shock tube under subsonic flow conditions. In these tests the flow conditions about the model were such that the heat transfer rate could be easily calculated, and the time scale of the measurement was similar to that obtained in the hypersonic shock tunnel. Heat transfer records both at the stagnation point of blunt bodies and on flat plate airfoils placed in the center of the shock tube were taken. In all cases the agreement between the calculated temperature rise and the temperature rise measured by the gage has been excellent. Fig. 12 is a record obtained on a flat plate airfoil on which a suddenly generated Mach 0.3 flow was obtained in a shock tube. The dotted line is the calculated theoretical temperature rise under these operating conditions, and the heavy solid line is the oscilloscope trace. The fall-off at 4 millisecc is not due to the gage but shows the arrival of secondary waves which change the flow conditions.

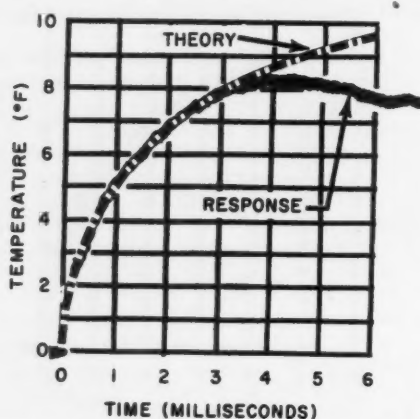


Fig. 12 Response of a resistance thermometer to a temperature step of 200 F with the theoretical response superimposed

Also, at the present time techniques of instrumentation of other aerodynamic parameters, such as force and pressure measurements, are being explored. The preliminary results of this program have indicated that the measurement of model forces by means of a special balance system is possible within the available testing time of the hypersonic shock tunnel.

Special Applications of the Shock Tube

As pointed out in the introduction, information about the nonequilibrium properties of air is required for the study of hypersonic flight. While much information may be obtained from the conventional shock tube and the hypersonic shock tunnel, shock tubes of special design may be required for some phases.

For example, a special modification of the shock tube may be employed to investigate high temperature chemical reactions, especially those which occur in high temperature air. Fig. 13 shows the modification of the conventional shock tube employed at Cornell Aeronautical Laboratory to study chemical kinetics (17). This "chemical" shock tube consists of a central driver section containing inert gas at high pressure, separated by diaphragms from the section containing the reactant gas and a large evacuated damping chamber. When the diaphragm between the driver gas and the reactant is ruptured, a shock wave travels through the reactant mixture heating and compressing it. This shock wave is then reflected at the end of the reactant section, further compressing and heating the reactant gas to the final reaction conditions. The rapid travel of the reflected shock wave through the re-

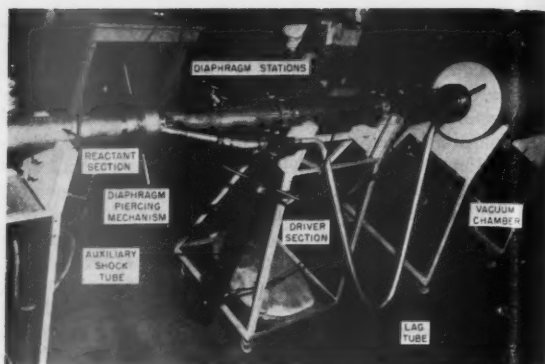


Fig. 13 Cornell Aeronautical Laboratory chemical shock tube

actant gas establishes a uniform condition of pressure and temperature for times on the order of several milliseconds. At a predetermined time, the diaphragm separating the driver from the evacuated chamber is burst. The expansion fan generated by the sudden flow of driver gas into the vacuum chamber propagates down the shock tube through the reacting mixture, cooling and expanding the reacting gas. In the chemical shock tube at Cornell Aeronautical Laboratory it is estimated that cooling rates on the order of five hundred thousand degrees Kelvin per second are obtained. In addition, the large evacuated tank acts as a damper, preventing waves from reflecting back into the reaction zone so that the reactant mixture is subjected to a single temperature pulse.

The first reaction to be studied in the chemical shock tube was the fixation of atmospheric nitrogen. Fig. 14 shows the results of some recent measurements of nitric oxide formation from air. It may be observed that equilibrium amounts of nitric oxide were not achieved in the low temperature region of the curve because the reaction rates and heating times are small. At higher temperatures, the reaction rates are large, and equilibrium amounts of nitric oxide were formed during the short heating times. Nevertheless, some of the nitric oxide decomposes during the cooling phase of the pulse since the cooling rates are inadequate to "freeze" the high temperature equilibrium. From a study of the approach to equilibrium, it is possible to determine the kinetics of the reaction.

One technique that has been employed in the chemical shock tube which may be of significance for other shock tube studies is the use of a "tailored interface." By adjusting the composition of the driver and the reactant gases, the shock reflected from the closed end of the tube will be transmitted without reflection through the interface. Since no reflection from the interface occurs, a constant pressure and temperature condition can be maintained in the reaction zone for a relatively long period of time (8).

The tailored interface technique may be used for aerodynamic testing since it may enable longer testing times

(Continued on page 568)

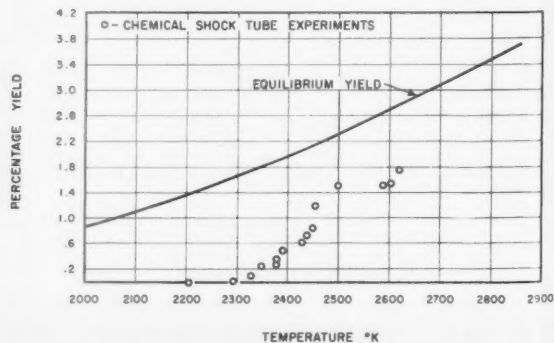


Fig. 14 Yield of nitric oxide from air at elevated temperatures

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Transfer Between Circular Orbits

D. F. LAWDEN¹

Canterbury University College, Christchurch, New Zealand

A solution is given to the problem of transferring a rocket from a circular orbit about one planet into another about a second planet with minimum expenditure of fuel. The planetary orbits are assumed to be coplanar and the longitudes of the planets in their orbits at the instants of departure and arrival of the rocket are supposed to be specified. The case of transfer between the Earth and Mars is taken as a numerical example of the general theory.

1 The Problem

A METHOD of calculating the orbit along which a rocket may be transferred from a circular orbit about one planet into another about a second planet with a minimum fuel expenditure has been described in another paper (1).² In addition to specifying the optimal trajectory of transfer, the method also determines the most favorable positions of the planets in their orbits about the Sun at the instant of departure. It may be necessary to delay departure for some considerable time until these favorable positions are attained by both planets. If, however, such a delay is not acceptable, the problem arises of calculating the most satisfactory mode of transfer when the two planets are in given relative positions on the chosen date of departure. This is the problem we shall solve in this paper.

2 Solution to the Problem

We shall restrict ourselves to the two-dimensional problem of transfer between two planets moving in coplanar elliptical orbits about the Sun. Ox, Oy are fixed rectangular Cartesian axes in the plane of the orbits. The method may be generalized to deal with the realistic three-dimensional problem by introducing into the argument a third axis Oz . However, the principles upon which the solution to this more complex problem is based will be identical in character with those which lead to a solution of the more elementary one, and no advantage will be gained, therefore, at this stage, by further complicating the analysis in order to achieve complete generality. Again, absolute accuracy could only be achieved by taking into account the perturbations caused in the planets' motions by their mutual attraction and the attractions of other bodies of the solar system, but the saving in fuel which would result would be negligible by comparison with the inevitable losses which will have to be accepted on account of our inability to navigate the rocket along any prescribed track without error. The most satisfactory procedure will always be to disregard all such small effects when computing an optimal track and then to allow for these, and earlier errors of navigation, during passage, by small correctional thrusts from the motor applied at various check points along the transfer orbit. A method of computing such corrections has been outlined in (2).

As in (1), we shall suppose that the rocket escapes from its circular orbit about the planet of departure by means of an

impulsive thrust from its motor directed along the tangent to this orbit. It then recedes along a hyperbolic trajectory, merging imperceptibly with the elliptical orbit of transfer in which it moves under the influence of the Sun alone. Coming under the influence of the planet of arrival, it falls along a hyperbolic track whose apse is at the level of the circular orbit. Upon arrival at the apse, it is transferred into the circular orbit by means of a second impulsive thrust in a direction opposing its motion, i.e., tangential to the circular orbit. Thus, apart from two short periods of thrust, the motor is inoperative.

If $(-f, -g)$ represent the x and y components of the gravitational field intensity acting upon the rocket when at the point (x, y) at time t , both f and g will be functions of the variables (x, y, t) . Let $x = X(t)$, $y = Y(t)$ be the equations defining the optimal trajectory of the rocket. Along this trajectory, we define a vector called the *primer* having components (u, v) satisfying the equations

$$\left. \begin{aligned} \ddot{u} + u \frac{\partial f}{\partial X} + v \frac{\partial g}{\partial X} &= 0 \\ \ddot{v} + u \frac{\partial f}{\partial Y} + v \frac{\partial g}{\partial Y} &= 0 \end{aligned} \right\} \dots\dots\dots [1]$$

It is shown in (1) and (3) that over an absolute optimal track, all motor thrusts must be impulsive in character and, in addition, the following conditions must be satisfied:

- (a) $u^2 + v^2 \leq 1$ at all points.
- (b) (u, v) are the direction cosines of the direction of thrust at each junction at which the motor operates.
- (c) (u, v) are continuous over such a junction.
- (d) $A = uf + vg + X\dot{u} + Y\dot{v}$ is continuous across a junction.
- (e) $A = 0$ over that section of the orbit of transfer where the attraction of the planets is negligible.

The above conditions were obtained on the assumption that we were free to choose the junctions at which impulsive thrusts were to be generated to suit ourselves. In the case under consideration, this is not so, and a reconsideration of the argument of (3) will show that, if the position of any junction is fixed at the outset, condition (c) may be waived at this junction. We shall not, therefore, require this condition to be satisfied at either of the junctions of the present problem.

Over the hyperbolic trajectory of departure from the initial circular orbit, we shall neglect the effect of the Sun's attraction. Let μ_0/r_0^2 be the attraction per unit mass of the planet of departure at a distance r_0 from its center. Let (e_0, l_0) be the eccentricity and semi-latus rectum, respectively, of the hyperbolic orbit. Its polar equation is then

$$\frac{l_0}{r_0} = 1 + e_0 \cos \psi_0 \dots\dots\dots [2]$$

where ψ_0 is measured from the apse. If (U_0, V_0) are the components of the primer in the directions of the radius vector r_0 and of the perpendicular to it, respectively, it is proved in (4) that the appropriate solution of Equations [1] is

$$U_0 = P_0 \cos \psi_0 + (Q_0 + H_0 l_0) e_0 \sin \psi_0 \dots\dots\dots [3]$$

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¹ Professor of Mathematics. Member of the British Interplanetary Society.

² Numbers in parentheses indicate References at end of paper.

$$V_0 = -P_0 \sin \psi_0 + (Q_0 + H_0 I_0) (1 + e_0 \cos \psi_0) + \frac{R_0 - P_0 \sin \psi_0 + H_0 \cot \psi_0}{1 + e_0 \cos \psi_0} \quad [4]$$

where P_0, Q_0, R_0, H_0 are dimensionless constants of integration and

$$I_0 = \frac{1}{2(e_0 - 1)^2} \tan^{1/2} \psi_0 - \frac{1}{2(e_0 + 1)^2} \cot^{1/2} \psi_0 - \frac{6e_0^2}{(e_0^2 - 1)^{3/2}} \tanh^{-1} \left(\sqrt{\frac{e_0 - 1}{e_0 + 1}} \tan^{1/2} \psi_0 \right) + \frac{e_0^3}{(e_0^2 - 1)^2} \frac{\sin \psi_0}{1 + e_0 \cos \psi_0} \quad [5]$$

An impulsive thrust is applied at the apse in a direction at right angles to the radius vector. Hence, by condition (b) above, we must have $U_0 = 0, V_0 = 1$ at $\psi_0 = 0$. This leads to the conditions

$$P_0 - \frac{e_0}{(e_0 + 1)^2} H_0 = 0 \quad [6]$$

$$Q_0(1 + e_0) + \frac{R_0}{1 + e_0} = 1 \quad [7]$$

It should be noted that $\psi_0 = 0$ makes the expression for I_0 indeterminate. We must accordingly let $\psi_0 \rightarrow 0$ in Equation [5].

Let μ/r^2 be the attraction per unit mass of the Sun at a distance r from its center. Let the polar equation of the transfer trajectory, over which the attractions of the planets are negligible, be

$$\frac{l}{r} = 1 + e \cos \psi \quad [8]$$

where ψ is measured from perihelion. If (U, V) are the components of the primer in directions along and perpendicular to r , respectively, then, as shown in (4), since $A = 0$ (condition (e))

$$U = P \cos \psi + Q e \sin \psi \quad [9]$$

$$V = -P \sin \psi + Q(1 + e \cos \psi) + \frac{R - P \sin \psi}{1 + e \cos \psi} \quad [10]$$

where P, Q, R are constants of integration to be determined later. These equations must be identical with Equations [3] and [4] where the hyperbolic trajectory merges with the transfer orbit. The primer components being solutions of second order differential Equations [1], we can ensure this by equating the values of U_0, V_0 and their first derivatives obtained from Equations [3] and [4] with the corresponding quantities obtained from Equations [9] and [10].

If ϕ_0 is the angle made by the direction of the asymptote of the hyperbolic trajectory of departure with the perpendicular to the radius vector r drawn in the sense of ψ increasing, it is shown in (1) that the components of the time derivative of the primer vector as computed from Equations [9] and [10] are

$$(S \sin \phi_0 + T \cos \phi_0) \dot{\psi} \text{ along the asymptote} \quad [11]$$

$$(-S \cos \phi_0 + T \sin \phi_0) \dot{\psi} \text{ perpendicular to the asymptote} \quad [12]$$

where

$$S = \frac{P \sin \psi - R}{1 + e \cos \psi} - Q \quad [13]$$

$$T = \frac{Re \sin \psi - Pe - P \cos \psi}{(1 + e \cos \psi)^2} \quad [14]$$

Consider now the form taken by Equations [3] and [4] at a great distance along the asymptote of the hyperbolic orbit.

At such a point, $r_0 \rightarrow \infty$ and hence $\cos \psi_0 = -1/e_1$ and $\sin \psi_0 = (e_0^2 - 1)^{1/2}/e_0$. Thus ψ_0 does not vary with t and hence

$$\dot{U}_0 = H_0 I_0 e_0 \sin \psi_0 = H_0 (e_0^2 - 1)^{1/2} I_0 \quad [15]$$

The first two terms in expression [5] for I_0 are constant and hence make no contribution to \dot{I}_0 . The last two terms become larger however, and hence must be dealt with separately. Differentiating these two terms with respect to t , we obtain

$$\frac{e_0^2 (e_0^2 - 3 - 2e_0 \cos \psi_0)}{(e_0^2 - 1)^2 (1 + e_0 \cos \psi_0)^2} \dot{\psi}_0 \quad [16]$$

Since $\cos \psi_0 = -1/e_0$, and in view of Equation [2], this may be written

$$\frac{e_0^2}{(e_0^2 - 1) l_0^2} r_0^2 \dot{\psi}_0 \quad [17]$$

But $r_0^2 \dot{\psi}_0$ is the constant velocity moment of the rocket about the center of attraction and is accordingly equal to $(\mu_0 l_0)^{1/2}$. Thus

$$\dot{I}_0 = \frac{e_0^2 \mu_0^{1/2}}{(e_0^2 - 1) l_0^{3/2}} \quad [18]$$

It now follows that, at a great distance from the planet of departure

$$\dot{I}_0 = H_0 e_0^2 \sqrt{\frac{\mu_0}{(e_0^2 - 1) l_0^3}} \quad [19]$$

Now consider Equation [4]. It may be shown that as $\cos \psi \rightarrow -1/e_0, I_0(1 + e_0 \cos \psi) \rightarrow$ a constant. It follows that the only term which makes a nonzero contribution to \dot{V}_0 when r_0 is large is

$$\frac{R_0 - P_0 \sin \psi_0 + H_0 \cot \psi_0}{1 + e_0 \cos \psi_0} \quad [20]$$

Referring to Equation [2], we see that this may be written in the form

$$\frac{1}{l_0} (R_0 - P_0 \sin \psi_0 + H_0 \cot \psi_0) r_0 = \frac{1}{l_0} \left[R_0 - P_0 \frac{(e_0^2 - 1)^{1/2}}{e_0} - H_0 \frac{1}{(e_0^2 - 1)^{1/2}} \right] r_0 \quad [21]$$

when $\psi_0 = \cos^{-1}(-1/e_0)$. Differentiating with respect to the time, we obtain

$$\dot{V}_0 = \frac{1}{l_0} \left[R_0 - P_0 \frac{(e_0^2 - 1)^{1/2}}{e_0} - H_0 \frac{1}{(e_0^2 - 1)^{1/2}} \right] \dot{r}_0 \quad [22]$$

But \dot{r}_0 is the velocity of recession from the planet of departure. Hence

$$\dot{r}_0 = [\mu_0 (e_0^2 - 1)/l_0]^{1/2}$$

and

$$\dot{V}_0 = \frac{\mu_0^{1/2}}{l_0^{3/2}} \left[R_0 (e_0^2 - 1)^{1/2} - P_0 \frac{e_0^2 - 1}{e_0} - H_0 \right] \quad [23]$$

Equations [19] and [23] give the components of the time derivative of the primer along and perpendicular to the asymptote of the hyperbolic orbit when the rocket has receded to a great distance from the planet of departure. Equating these quantities with the corresponding components given at [11] and [12], we obtain the equations

$$(S \sin \phi_0 + T \cos \phi_0) \frac{\dot{\psi} l_0^{3/2}}{\mu_0^{1/2}} = \frac{H_0 e_0^2}{(e_0^2 - 1)^{1/2}} \quad [24]$$

$$(-S \cos \phi_0 + T \sin \phi_0) \frac{\dot{\psi} l_0^{3/2}}{\mu_0^{1/2}} = R_0 (e_0^2 - 1)^{1/2} -$$

$$P_0 \frac{e_0^2 - 1}{e_0} - H_0 \quad [25]$$

But $\mu_0^{1/2}/l_0^{1/2} = (e_0 + 1)^{-1/2} \times$ angular velocity of the rocket in the circular orbit about the planet of departure and hence is large by comparison with ψ , the angular velocity of the rocket about the Sun at the commencement of the transfer orbit. We shall accordingly approximate Equations [24] and [25] by the equations

$$\frac{H_0 e_0^2}{(e_0^2 - 1)^{1/2}} = R_0(e_0^2 - 1)^{1/2} - P_0 \frac{e_0^2 - 1}{e_0} - H_0 = 0 \quad [26]$$

Solving these latter equations in conjunction with Equations [6] and [7], we obtain

$$P_0 = R_0 = H_0 = 0 \quad Q_0 = \frac{1}{1 + e_0} \quad [27]$$

Thus, from Equations [3] and [4], at a great distance from the planet of departure

$$U_0 = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \quad V_0 = 0 \quad [28]$$

This implies that the components of the primer, along and perpendicular to the radius vector r from the Sun, at the commencement of the trajectory of transfer are

$$U = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \sin \phi_0 \quad V = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \cos \phi_0 \quad [29]$$

Identical results must be given by Equations [9] and [10] and thus

$$P \cos \psi + Qe \sin \psi = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \sin \phi_0 \quad [30]$$

$$-P \sin \psi + Q(1 + e \cos \psi) + \frac{R - P \sin \psi}{1 + e \cos \psi} = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \cos \phi_0 \quad [31]$$

where ψ takes the value appropriate to the commencement of the orbit of transfer.

The transition from the transfer orbit into the hyperbolic orbit of approach is dealt with similarly and yields the conditions

$$P \cos \psi' + Qe \sin \psi' = \sqrt{\frac{e_1 - 1}{e_1 + 1}} \sin \phi_1 \quad [32]$$

$$-P \sin \psi' + Q(1 + e \cos \psi') + \frac{R - P \sin \psi'}{1 + e \cos \psi'} = \sqrt{\frac{e_1 - 1}{e_1 + 1}} \cos \phi_1 \quad [33]$$

ψ' being the value of ψ appropriate to the end of the transfer orbit, ϕ_1 being the angle made by the asymptote of the approach orbit with the perpendicular to the radius vector r , and e_1 being the eccentricity of the hyperbola of approach.

Condition (d), above, may be applied at each junction, but yields equations involving the constants of integration which appear when Equations [1] are integrated over the circular orbits about the planets. Such equations do not limit the trajectory of transfer in any way, but only serve to fix the new constants of integration.

Elimination of the quantities P, Q, R between Equations [30] to [33] now yields the condition to be satisfied by the optimal transfer orbit, viz.

$$\begin{aligned} &E_0 \sin \phi_0 [(2 + e \cos \psi) \cos (\psi - \psi') - (2 + e \cos \psi')] \\ &+ E_1 \sin \phi_1 [(2 + e \cos \psi') \cos (\psi - \psi') - (2 + e \cos \psi)] \\ &+ [E_0(1 + e \cos \psi) \cos \phi_0 - E_1(1 + e \cos \psi') \cos \phi_1] \times \\ &\sin (\psi' - \psi) = 0 \quad [34] \end{aligned}$$

where

$$E_0 = \sqrt{\frac{e_0 - 1}{e_0 + 1}} \quad E_1 = \sqrt{\frac{e_1 - 1}{e_1 + 1}} \quad [35]$$

Let (r_0, r_1) be the respective distances of the planets of departure and arrival from the Sun at the times of departure or arrival of the rocket, and let (θ_0, θ_1) be the respective longitudes of the planets at these instants. If γ is the longitude of perihelion on the transfer orbit, it then follows that at the terminals of the transfer orbit

$$\psi = \theta_0 - \gamma, \quad \psi' = \theta_1 - \gamma \quad [36]$$

and hence, from Equation [8], that

$$\frac{l}{r_0} = 1 + e \cos (\theta_0 - \gamma) \quad [37]$$

$$\frac{l}{r_1} = 1 + e \cos (\theta_1 - \gamma) \quad [38]$$

These latter equations permit us to write the condition [34] in the form

$$\begin{aligned} &\frac{E_1}{E_0} = \frac{\frac{l}{r_0} \sin (\theta_1 - \theta_0) \cos \phi_0 + \left[\left(1 + \frac{l}{r_0} \right) \cos (\theta_1 - \theta_0) - \left(1 + \frac{l}{r_1} \right) \right] \sin \phi_0}{\frac{l}{r_1} \sin (\theta_1 - \theta_0) \cos \phi_1 - \left[\left(1 + \frac{l}{r_1} \right) \cos (\theta_1 - \theta_0) - \left(1 + \frac{l}{r_0} \right) \right] \sin \phi_1} \quad [39] \end{aligned}$$

If w_0 is the velocity of the rocket in its circular orbit about the planet of departure and relative to this body, its velocity at infinity on the hyperbolic orbit along which it leaves the planet is $w_0(e_0 - 1)^{1/2}$. This velocity must equal the vector difference between the velocity of the planet in its orbit at the time of departure and the velocity of the rocket relative to the Sun at the commencement of the transfer orbit. If λ_0 is the semi-latus rectum, e_0 is the eccentricity, and γ_0 is the longitude of perihelion of the orbit of the planet of departure, it may be shown (5) that this leads to the conditions

$$w_0(e_0 - 1)^{1/2} \cos \phi_0 = \mu^{1/2} (l^{1/2} - \lambda_0^{1/2})/r_0 \quad [40]$$

$$w_0(e_0 - 1)^{1/2} \sin \phi_0 =$$

$$\mu^{1/2} \left[\frac{e}{l^{1/2}} \sin (\theta_0 - \gamma) - \frac{e_0}{\lambda_0^{1/2}} \sin (\theta_0 - \gamma_0) \right] \quad [41]$$

Whence, eliminating ϕ_0 , we obtain

$$\begin{aligned} w_0^2 (e_0 - 1) = \mu \left[\frac{e^2}{l} + \frac{e_0^2}{\lambda_0} - \frac{2e e_0}{l^{1/2} \lambda_0^{1/2}} \cos (\gamma - \gamma_0) - \right. \\ \left. \left(\frac{1}{l^{1/2}} - \frac{1}{\lambda_0^{1/2}} \right)^2 \left(1 + \frac{2l^{1/2} \gamma_0^{1/2}}{r_0} \right) \right] \quad [42] \end{aligned}$$

as explained in (5).

The following equations may be found similarly

$$w_1(e_1 - 1)^{1/2} \cos \phi_1 = \mu^{1/2} (\lambda_1^{1/2} - l^{1/2})/r_1 \quad [43]$$

$$w_1(e_1 - 1)^{1/2} \sin \phi_1 =$$

$$\mu^{1/2} \left[\frac{e_1}{\lambda_1^{1/2}} \sin (\theta_1 - \gamma_1) - \frac{e}{l^{1/2}} \sin (\theta_1 - \gamma) \right] \quad [44]$$

$$\begin{aligned} w_1^2 (e_1 - 1) = \mu \left[\frac{e^2}{l} + \frac{e_1^2}{\lambda_1} - \frac{2e e_1}{l^{1/2} \lambda_1^{1/2}} \cos (\gamma - \gamma_1) - \right. \\ \left. \left(\frac{1}{l^{1/2}} - \frac{1}{\lambda_1^{1/2}} \right)^2 \left(1 + \frac{2l^{1/2} \gamma_1^{1/2}}{r_1} \right) \right] \quad [45] \end{aligned}$$

Eliminating ϕ_0 and ϕ_1 from the condition [39] by the use

of Equations [40], [41], [43], and [44], we obtain it in the form

$$\sqrt{\frac{w_0^2(e_0 + 1)}{w_1^2(e_1 + 1)}} = \frac{L_0}{L_1} \dots \dots \dots [46]$$

where

$$\begin{aligned} L_0 &= \frac{l}{r_0^2} (l^{1/2} - \lambda_0^{1/2}) \sin(\theta_1 - \theta_0) \\ &+ \left[\left(1 + \frac{l}{r_0}\right) \cos(\theta_1 - \theta_0) - \left(1 + \frac{l}{r_1}\right) \right] \times \\ &\quad \left[\frac{e}{l^{1/2}} \sin(\theta_0 - \gamma) - \frac{e_0}{\lambda_0^{1/2}} \sin(\theta_0 - \gamma_0) \right] \\ L_1 &= \frac{l}{r_1^2} (\lambda_1^{1/2} - l^{1/2}) \sin(\theta_1 - \theta_0) \\ &- \left[\left(1 + \frac{l}{r_1}\right) \cos(\theta_1 - \theta_0) - \left(1 + \frac{l}{r_0}\right) \right] \times \\ &\quad \left[\frac{e_1}{\lambda_1^{1/2}} \sin(\theta_1 - \gamma_1) - \frac{e}{l^{1/2}} \sin(\theta_1 - \gamma) \right] \end{aligned}$$

Equations [37], [38], [42], [45], and [46] determine the five unknowns e_0 , e_1 , l , e , γ and hence the optimal mode of transfer. These equations are easily reduced to one condition, since Equations [42], [45] may be employed to eliminate e_0 and e_1 from Equation [46] and then, solving Equations [37], [38] for l and e in terms of γ , these quantities also may be eliminated. We are left with a single equation for γ which must be solved numerically.

3 Transfer Between Circular Orbits

The orbits of the principal bodies of the solar system being very nearly circular, the simplified forms taken by the equations of the last section when it is assumed that the orbits of the planets of departure and arrival are exactly circular are of some importance. In this case, $e_0 = e_1 = 0$ and λ_0, λ_1 are the radii of the planetary orbits. Also $r_0 = \lambda_0, r_1 = \lambda_1$. Thus Equations [42], [45] can now be written

$$w_0^2(e_0 - 1) = \mu \left(\frac{e^2 - 1}{l} + \frac{3}{r_0} - \frac{2l^{1/2}}{r_0^{3/2}} \right) \dots \dots [47]$$

$$w_1^2(e_1 - 1) = \mu \left(\frac{e^2 - 1}{l} + \frac{3}{r_1} - \frac{2l^{1/2}}{r_1^{3/2}} \right) \dots \dots [48]$$

Condition [46] is then equivalent to

$$\sqrt{\frac{e^2 - 1 + \frac{3l}{r_0} - 2\left(\frac{l}{r_0}\right)^{3/2} + \frac{2w_0^2}{\mu}l}{e^2 - 1 + \frac{3l}{r_1} - 2\left(\frac{l}{r_1}\right)^{3/2} + \frac{2w_1^2}{\mu}l}} = \frac{G_0}{G_1} \dots \dots [49]$$

where

$$\begin{aligned} G_0 &= \left[\left(\frac{l}{r_0}\right)^2 - \left(\frac{l}{r_0}\right)^{3/2} \right] \sin(\theta_1 - \theta_0) + \\ &\quad \left[\left(1 + \frac{l}{r_0}\right) \cos(\theta_1 - \theta_0) - 1 - \frac{l}{r_1} \right] e \sin(\theta_0 - \gamma) \\ G_1 &= \left[\left(\frac{l}{r_1}\right)^2 - \left(\frac{l}{r_1}\right)^{3/2} \right] \sin(\theta_0 - \theta_1) + \\ &\quad \left[\left(1 + \frac{l}{r_1}\right) \cos(\theta_0 - \theta_1) - 1 - \frac{l}{r_0} \right] e \sin(\theta_1 - \gamma). \end{aligned}$$

This latter equation, together with Equations [37] and [38], specify the optimal trajectory.

If the rocket is to be transferred from a circular orbit just outside the Earth's atmosphere into another close to the surface of Mars, the following values must be substituted into Equation [49]

$$\begin{aligned} r_0 &= 1.497 \times 10^{13}, & r_1 &= 2.280 \times 10^{13} \\ \mu &= 1.33 \times 10^{28}, & w_0 &= 7.912 \times 10^5, & w_1 &= 3.557 \times 10^5 \end{aligned}$$

the units being cgs. In this case, we have solved Equation [49] for γ when $(\theta_1 - \theta_0)$ takes the values shown in Table 1. The corresponding values of l and e are also given, together with the net characteristic velocity V (km/sec) of the maneuver and the time of transit T (days). The amount $\Delta\omega$ by which the longitude of Mars exceeds that of the Earth at the instant of departure is shown in the last column.

When $\theta_1 - \theta_0 = 180$ deg, the optimal transfer orbit is tangential to both the planetary orbits. This is the well-known "Hohmann" case and corresponds to an over-all optimum.

For values of $\Delta\omega$ other than those within the range of the table, the characteristic velocity is far too large to be practicable, or more than one complete circuit of the transfer orbit must be made so that the waiting time is greater than that which must elapse to bring Mars into a more favorable position. There will accordingly be periods during which transfer between the Earth and Mars is not practicable. This phenomenon has been discussed by Preston-Thomas (6).

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Table 1

$\theta_1 - \theta_0$	γ	l	e	V	T	$\Delta\omega$
45°	-108.5°	1.259×10^{13}	0.5010	18.58	80	3.2°
90°	-62.5°	1.696×10^{13}	0.2889	9.61	136	18.7°
135°	-30°	1.786×10^{13}	0.2241	6.38	198	31.5°
180°	0°	1.807×10^{13}	0.2076	5.75	259	44.4°
225°	30°	1.786×10^{13}	0.2241	6.38	317	58.8°
270°	62.5°	1.696×10^{13}	0.2889	9.61	366	78.2°
315°	108.5°	1.259×10^{13}	0.5010	18.58	354	129.5°

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Thermonuclear Power Plants

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Some of the unique features of thermonuclear power plants and the essential problems in the technical design of such plants are discussed in this paper. The thermonuclear reaction rate for the fusion of deuterium is calculated on the basis of a similar analysis published by Gamow and Teller. The pressure, temperature, and minimum dimensions of the necessary reaction chamber are determined largely by consideration of reaction quenching and energy loss near the walls. Results are presented for the power output and the efficiency of a power station utilizing the deuterium fusion reaction. The comment by Greenstein that follows this paper deals particularly with the difficult problem of calculating the reaction quenching and energy loss rates at the walls.

1 Introduction

A GREAT majority of the present discussions and plans on the utilization of nuclear energy for power production is related to the fission reaction. Although the nuclear fission power plants have many distinct advantages over the conventional power plants, the limited world reserve of uranium and thorium that can be practically mined makes the long term prospect of such power plants somewhat uncertain. On the other hand, the thermonuclear fusion reaction, particularly the "burning" of deuterium into helium, utilizes a very abundant fuel. Therefore, if the fusion reaction can be made to generate energy for power plants, the prospect of world energy supply will be very much brighter.

But can the fusion reaction be utilized in terrestrial power plants? This question has been examined by E. Sanger and his collaborators (1, 2).² However, a critical reading of Sanger's work will show that part of his analysis is not valid because he has not gone deep enough into the subject. It is the purpose of this note to point out some of the unique features of the thermonuclear power plants and the essential problems in the technical design of such power plants. It will be seen that such engineering projects are truly of stupendous proportions and are a challenge to one's imagination. However, the reward to the welfare of the human race by a successful development of thermonuclear power plants is so great as to make the careful examination of this problem a very worthwhile research project.

2 Thermonuclear Reaction Rate

Thermonuclear reactions are reactions between charged nuclei. Because of the electric charge, necessarily positive, the approach of the nuclei to each other has to overcome the Coulomb repulsion between the nuclei. Therefore, only if the relative kinetic energy of the nuclei is high, can a close enough approach be obtained and a reaction take place. This required kinetic energy is so large that even at temperatures as high as 10^8 K only nuclei in the high energy end of the Maxwellian velocity distribution can achieve reaction. Therefore only a very small fraction of the nuclei participate in the reaction. In other words, the reaction rate is quite small. This observation leads to a great simplification of the calculation:

The nuclear distribution can be considered as quasi-steady; i.e., the Maxwellian distribution can in fact be used in spite of the slight lack of thermodynamic equilibrium during the reaction. Gamow and Teller (3) have developed such a theory of thermonuclear reactions. The following is a slightly modified form of the theory suitable for the present purpose.

Considering the colliding particles as rigid spheres, it is well known (4) that the number dN of collisions per unit volume per unit time between particles of type 1 and 2 with kinetic energy of collision between ϵ and $\epsilon + d\epsilon$ is

$$dN = \frac{n_1 n_2}{s} \left(\frac{2\pi kT}{\mu} \right)^{1/2} e^{-\epsilon/kT} 2D_{12}^2 \frac{\epsilon}{kT} d\epsilon \dots [1]$$

where n_1, n_2 are the number densities of particles of type 1 and 2, respectively; s is the symmetry number, equal to 2 if type 1 and type 2 are the same, and equal to 1 if not; μ is the reduced mass of type 1 and type 2 particles. D_{12} is the average diameter of the two types of particles; i.e., if D_1 and D_2 are the diameters of type 1 and 2 particles, then

$$D_{12} = \frac{1}{2} (D_1 + D_2) \dots [2]$$

To put this general formula into a form more useful for the present computation, molar fractions ν_1 and ν_2 are introduced

$$\nu_1 = \frac{n_1}{n} \quad \nu_2 = \frac{n_2}{n} \quad n = \sum n_i \dots [3]$$

where n is the total number of particles per unit volume; the particles may include electrons besides the nuclei. Furthermore, if M_1 and M_2 are the mass of particles of type 1 and 2, and A_1 and A_2 are the corresponding quantities expressed in terms of atomic mass units ("atomic weights" of the nuclei), and M is the mass of one atomic mass unit, then

$$\mu = \frac{M_1 M_2}{M_1 + M_2} = \frac{A_1 A_2}{A_1 + A_2} M = AM \dots [4]$$

Thus A is the reduced mass expressed in terms of atomic mass units. If V is the relative velocity of the two colliding particles, then ϵ , the relative translational energy, is defined as

$$\epsilon = \frac{1}{2} \mu V^2 \dots [5]$$

If P is the thermodynamic pressure, then

$$n = P/kT \dots [6]$$

Equation [6] is true only if the assembly of particles is at thermodynamic equilibrium and if the particles essentially do not interact, i.e., the assembly is a perfect gas. At the extremely low gas density that will be considered, this is true to a high degree of accuracy. Of course, if the sum of the particles is not at thermodynamic equilibrium, e.g., fusion product neutrons which hardly collide a sufficient number of times with other particles to have a Maxwellian distribution of velocity within the dimension of region considered, then such particles must not be considered in calculating the "total" particle density n . The "pressure" produced by such particles on the wall of the containing vessel has to be treated separately. On the other hand, photons, if any, that are almost at thermodynamic equilibrium must be included in the particle density n .

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² Numbers in parentheses indicate References at end of paper.

The most important modification of [1] is the following: Nuclear reactions cannot be considered as collisions between rigid spheres but are expressed through an effective cross section σ . The effective cross section for collision of rigid spheres is a circle of radius D_{12} . Therefore

$$D_{12}^2 = \sigma/\pi. \quad [7]$$

By substituting [3], [4], [6], and [7] into [1], the number dN is

$$dN = \frac{4\pi v_1 v_2 \sigma P^2}{s\sqrt{2\pi AM}} \frac{e^{-\epsilon/kT}}{(kT)^{3/2}} \epsilon d\epsilon. \quad [8]$$

It is important to note that in general the effective cross section is not a constant but function of energy ϵ . In particular, according to the theory of nuclear reaction, the quantum penetration of Coulomb barrier gives the effective cross section σ as

$$\sigma = \frac{\Lambda^2}{4\pi} \frac{\Gamma AM R^2}{\hbar^2} \times \exp \left[-\frac{2\pi e^2 Z_1 Z_2 \sqrt{AM}}{\hbar \sqrt{2\epsilon}} + \frac{4e\sqrt{2AMZ_1 Z_2 R}}{\hbar} \right] \dots [9]$$

where Λ is the de Broglie wavelength

$$\Lambda = \frac{2\pi\hbar}{\sqrt{2AM\epsilon}} \quad [10]$$

R is the radius of the compound nucleus formed during reaction, and can be estimated as

$$R = [1.7 + 1.22 (A_1 + A_2)^{1/3}] \times 10^{-13} \text{ cm.} \quad [11]$$

Γ is the half-width of the nuclear resonance level. Z_1 and Z_2 are the nuclear charges in units of electronic charge e .

By substituting [9] and [10] into [8], it is seen that the variation of $dN/d\epsilon$ with respect to ϵ is due to the exponential factor

$$\exp \left\{ -\left[\frac{\epsilon}{kT} + \frac{2\pi e^2 Z_1 Z_2 \sqrt{AM}}{\hbar \sqrt{2\epsilon}} \right] \right\} \dots [12]$$

There is a minimum of the quantity within the square bracket with respect to ϵ which corresponds to the maximum of $dN/d\epsilon$. If this ϵ is denoted by ϵ^* , then

$$\frac{1}{kT} = \frac{\pi e^2 Z_1 Z_2 \sqrt{AM}}{\sqrt{2} \hbar \epsilon^{3/2}}$$

or

$$\epsilon^* = \left\{ \frac{\pi e^2 Z_1 Z_2 \sqrt{AM} kT}{\sqrt{2} \hbar} \right\}^{2/3} \quad [13]$$

Near this value of ϵ , the expression in the square bracket of [12] can be approximated by

$$\frac{\epsilon}{kT} + \frac{2\pi e^2 Z_1 Z_2 \sqrt{AM}}{\hbar \sqrt{2\epsilon}} \cong \frac{\epsilon^*}{kT} + \frac{2\pi e^2 Z_1 Z_2 \sqrt{AM}}{\sqrt{2} \hbar \sqrt{\epsilon^*}} + \frac{1}{2} \cdot \frac{3}{2} \times \frac{\pi e^2 Z_1 Z_2 \sqrt{AM}}{\sqrt{2} \hbar \epsilon^{5/2}} (\epsilon - \epsilon^*)^2 = 3 \frac{\epsilon^*}{kT} + \frac{3}{4} \frac{1}{kT \epsilon^*} (\epsilon - \epsilon^*)^2. \quad [14]$$

Thus, under this approximation, dN can be computed as

$$dN = \frac{4\pi v_1 v_2 \Gamma P^2}{s\sqrt{2\pi AM} (kT)^{3/2}} \frac{\pi R^2}{2} \exp \left[\frac{4e\sqrt{2AMZ_1 Z_2 R}}{\hbar} - \frac{3\epsilon^*}{kT} \right] \times \exp \left[-\frac{3}{4} \frac{1}{kT \epsilon^*} (\epsilon - \epsilon^*)^2 \right] d\epsilon. \quad [15]$$

By integrating over all ϵ , we have the number N of effective

binary collisions per unit vol per unit time as

$$N = \frac{4\pi v_1 v_2 R^2}{s\sqrt{3}} \left(\frac{P}{kT} \right)^2 \left(\frac{\Gamma}{kT} \right) \sqrt{\frac{kT}{2AM}} \sqrt{\frac{\epsilon^*}{kT}} \times \exp \left[\frac{4e\sqrt{2AMZ_1 Z_2 R}}{\hbar} - 3 \frac{\epsilon^*}{kT} \right]. \quad [16]$$

From [13]

$$\frac{\epsilon^*}{kT} = \left(\frac{\pi^2 e^4 Z_1^2 Z_2^2 AM}{2\hbar^2 kT} \right)^{1/3} \quad [17]$$

Equations [16] and [17] together determine the reaction rate. There is only one important difference between [16] and the original formula due to Gamow and Teller (3); Gamow and Teller have not included the symmetry number s and thus may be wrong in some cases by a factor of 2.

If we denote by x the quantity $(kT)^{1/3}$, then the temperature dependent part of [16] can be written as

$$x^3 \exp \left[-3 \left(\frac{\pi^2 e^4 Z_1^2 Z_2^2}{2\hbar^2} \right)^{1/3} x \right] \quad [18]$$

This quantity clearly has a maximum at some value of x , say x_0 ; x_0 is determined by

$$8 - 3 \left(\frac{\pi^2 e^4 AM Z_1^2 Z_2^2}{2\hbar^2} \right)^{1/3} x_0 = 0 \quad [19]$$

Equation [19] gives the optimum reaction temperature T_0 for maximum reaction rate at constant pressure as

$$T_0 = \left(\frac{3}{8} \right)^3 \frac{\pi^2 e^4 AM Z_1^2 Z_2^2}{2\hbar^2} \quad [20]$$

By putting in the numerical values of physical constants

$$T_0 = 1.442 \times 10^8 A Z_1^2 Z_2^2 \text{ } ^\circ\text{K.} \quad [21]$$

Equations [20] and [21] show that the optimum reaction temperature depends only on the reduced mass and charges of the nuclei and is independent of the details of the reaction. T_0 is the smallest for proton-proton reaction, ($A = 1$, $Z_1 = Z_2 = 1$), for which $T_0 = 0.721 \times 10^8$ K.

The important parameter in the expression of reaction rate is the level width Γ . This has to be determined experimentally. However the experimental reaction cross sections are usually expressed as

$$\sigma = \frac{B}{\epsilon} e^{-C/\sqrt{\epsilon}} \quad [22]$$

where B and C are two empirically determined constants for any one reaction. Same as the preceding paragraphs, [8] and [22] can be combined to give the formulas

$$\frac{\epsilon^*}{kT} = \left(\frac{C^2}{4kT} \right)^{1/3} \quad [23]$$

$$N = \frac{4\pi v_1 v_2}{s} \left(\frac{B}{kT} \right) \left(\frac{P}{kT} \right)^2 \sqrt{\frac{2kT}{3AM}} \sqrt{\frac{\epsilon^*}{kT}} e^{-3(\epsilon^*/kT)} \quad [24]$$

and

$$T_0 = \left(\frac{3}{8} \right)^3 \frac{C^2}{4k} \quad [25]$$

If the optimum temperature T_0 is used as a reference temperature

$$\frac{\epsilon^*}{kT} = \frac{8}{3} \left(\frac{T_0}{T} \right)^{1/3} \quad [26]$$

For numerical computation, P is usually given in atmospheres. Thus

$$\frac{P}{kT} = 7.34 \times 10^{21} \frac{P}{T} \text{ cm}^{-3} \dots\dots\dots [27]$$

B is usually given in units of barns-kilovolts, thus

$$\frac{B}{kT} = 1.160 \times 10^{-17} \frac{B}{T} \text{ cm}^2 \dots\dots\dots [28]$$

C is usually given in units of $\text{kv}^{1/2}$, thus

$$T_0 = 1.528 \times 10^5 C^2 \text{ } ^\circ\text{K} \dots\dots\dots [29]$$

If E is the energy production of a single binary reaction, then the rate of energy production Q per unit vol per unit time is obviously

$$Q = EN \dots\dots\dots [30]$$

3 Example: Deuterium Reaction

Because of the abundance of deuterium as a naturally occurring stable isotope of hydrogen, it is of interest to consider the burning of deuterium. The accurate reaction data were given recently by Arnold et al. (5)

$$\sigma = \frac{B'}{\epsilon'} e^{-C'/\sqrt{\epsilon'}} \dots\dots\dots [31]$$

where ϵ' is the deuteron energy in kilovolts in the usual laboratory coordinate system, and $B' = 288$ barns-kv, $C' = 45.7 \text{ kv}^{1/2}$. Since the ϵ in [22] is the relative kinetic energy defined by (5) and the ratio of deuterium mass and the reduced mass is 2 in a deuteron-deuteron reaction,

$$\epsilon' = 2\epsilon \dots\dots\dots [32]$$

Therefore, by using Arnold's data, the reaction constants B and C are

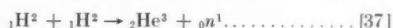
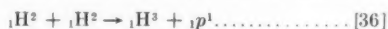
$$B = B'/2 = 144 \text{ barns-kv} \dots\dots\dots [33]$$

$$C = C'/\sqrt{2} = 32.36 \text{ kv}^{1/2} \dots\dots\dots [34]$$

Equation [29] immediately gives the optimum temperature T_0 for deuteron-deuteron reaction as

$$T_0 = 1.600 \times 10^5 \text{ } ^\circ\text{K} \dots\dots\dots [35]$$

According to Arnold et al., the deuteron-deuteron reaction branches, with almost equal probability, into two reactions



Since the masses of the atomic species are given as follows

$$\left. \begin{aligned} A({}_1\text{H}^2) &= 2.014735 \\ A({}_1\text{H}^3) &= 3.016997 \\ A({}_2\text{He}^3) &= 3.016977 \\ A({}_1\text{p}^1) &= 1.008142 \\ A({}_0\text{n}^1) &= 1.008982 \end{aligned} \right\} \dots\dots\dots [38]$$

The reaction [36] then produces

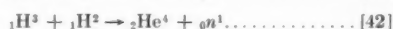
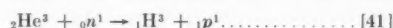
$$\begin{aligned} 2 \times 2.014735 - (3.016997 + 1.008142) &= 0.04331 \text{ amu} = \\ 4.03 \text{ Mev} &= 6.46 \times 10^{-6} \text{ ergs} \dots\dots [39] \end{aligned}$$

The reaction [37] produces

$$\begin{aligned} 2 \times 2.014735 - (1.008982 + 3.016177) &= 0.003511 \text{ amu} = \\ 3.27 \text{ Mev} &= 5.24 \times 10^{-6} \text{ ergs} \dots\dots [40] \end{aligned}$$

However, in a thermonuclear reaction chamber, [36] and

[37] do not represent the end of reactions. The reaction products are immediately thermalized by elastic collisions with other particles; then the following reactions are possible



At first sight it seems that the reaction [41] depending upon both components of the reaction products of low concentration will be very much less frequent than [42] and [43] depending upon only one component of reaction product. However at thermal energies corresponding to temperatures of the order of 10^5 K , the cross section of [41] is of the order of 10 barns, while according to Arnold (5), the reaction cross sections of [42] and [43] are very much smaller. Therefore [41] is in fact the only reaction of importance while [42] and [43] can be neglected. This being the case, then eventually all of the deuteron-deuteron reaction really results in the production of tritium according to [36]. Therefore the average energy production E for every deuteron-deuteron reaction is according to [39]

$$E = 6.46 \times 10^{-6} \text{ ergs} \dots\dots\dots [44]$$

The energy produced by burning a unit mass of deuterium is thus

$$\begin{aligned} \frac{6.46 \times 10^{-6} \times 2.388 \times 10^{-8} \times 0.605 \times 10^{24}}{4.029} \text{ cal/gr} = \\ 2.31 \times 10^{10} \text{ cal/gr} \dots\dots [45] \end{aligned}$$

Now assume $T = T_0 = 1.600 \times 10^5 \text{ K}$, $P = 100 \text{ atm}$. The feed gas D_2 will be completely ionized at this temperature, and reacting mixture will be composed initially of equal numbers of deuterons and electrons. Thus $\nu_1 = \nu_2 = 1/2$, $s = 2$. Then using [24], [26], [30] the energy production Q per unit vol per unit time is

$$\begin{aligned} Q &= 6.46 \times 10^{-6} \times \frac{1}{2} \times \frac{1.160 \times 10^{-17} \times 144}{1.600 \times 10^5} \times \\ &\left(\frac{7.34 \times 10^{23}}{1.600 \times 10^5} \right)^2 \sqrt{\frac{8}{3}} \times \frac{2 \times 8.316 \times 10^7 \times 1.6 \times 10^8}{3 \times 1.007} \times \\ &e^{-8} \text{ ergs/cm}^3 \text{ sec} = 0.365 \times 10^8 \text{ ergs/cm}^3 \text{ sec} = \\ &3.65 \text{ watts/cm}^3 = 0.874 \text{ cal/cm}^3 \text{ sec} \dots\dots [46] \end{aligned}$$

It is interesting to note that this volume rate of energy generation is only $1/10$ of the rate of generation as in a modern aircraft gas turbine combustion chamber using hydrocarbon fuel. Therefore in spite of the extremely high temperature, thermonuclear reaction using deuterium is a relatively slow reaction. The reason for this anomaly is the extremely low density of the hot gas: There simply are not enough deuterons in a unit volume to give high reaction rate. However, as Sanger (1) has shown, other nuclei generally give even lower rates of energy production.

4 Thermonuclear Reaction Chamber

The moderate volume rate of energy production together with the extremely high gas temperature naturally call ones attention to the problem of quenching of the "flame" by excessive cooling. This problem is in fact the central problem of thermonuclear reaction chamber. There is certainly a critical size, say a critical diameter, of the reaction chamber below which the reaction cannot be maintained. As a very rough first estimate, one may take the chemical combustion as a model, and use the mean free path as the sizing length. Because of the relatively slow thermonuclear reaction, the chemical model should be one of poor reactivity. Thus the quenching diameter at atmospheric pressure can be taken as

1 cm. The pressure effect on quenching can be thought of as a Reynolds number effect. Then the quenching diameter at 100 atm will be $1/100$ cm. To translate this value to thermonuclear reaction chamber, one notes the fact that the ratio of mean free path for the two cases is approximately 10^6 . Thus the rough estimate of the critical diameter D_c of the reaction chamber is

$$D_c = \frac{1}{100} \times 10^6 \text{ cm} = 100 \text{ meters} \dots\dots\dots [47]$$

If the length is to be ten times the diameter, then the thermonuclear reaction chamber is a vessel of 100 meters diam and 1000 meters long built to withstand a pressure of 100 atmospheres!

To examine the quenching problem in some detail, one must first estimate the mean free path of the fully ionized mixture of deuteron and electron. If n is the particle density, and if σ_s is the scattering cross section of the particles, then the general equation for the mean free path l is

$$l = \frac{0.177}{n\sigma_s} = \frac{0.177kT}{P\sigma_s} = 2.41 \times 10^{-23} \frac{T}{P\sigma_s} \dots\dots\dots [48]$$

For fully ionized particles, the cross section σ_s can be computed approximately according to Lin, Resler, and Kantrowitz (6) as

$$\sigma_s = 8.10 \left(\frac{e^2}{3kT} \right)^2 \log \left(\frac{kT}{e \sqrt{4\pi P}} \frac{e^2}{3kT} \right) \dots\dots\dots [49]$$

By taking $T = 10^8$ K, σ_s from [49] is equal to 4.00×10^{-20} cm². Then [48] gives a mean free path of $l = 603$ cm. With such a large free path, the transfer of energy by collisions is extremely slow and inefficient. To improve the chances of collision, some particles of larger size must be introduced, e.g., atoms of heavier elements. The heavier atoms can have their outer electrons stripped (ionized) at the prevailing high temperature, but since some electrons remain attached to the nucleus, the size of the partially ionized atom can still be of the order of A . Then such particles will be a scattering cross section of the order of 10^{-16} cm². Even with only one per cent of such heavier elements in the mixture, the mean free path will be brought down to a few centimeters. This is indeed the mean free path used in the size estimate of the preceding paragraph. Needless to say, the heavier atoms introduced must not capture neutrons appreciably so as not to interfere with the very important energy producing reaction of [41].

However, even with the presence of heavy partially ionized atoms, the mixture will be still practically transparent to high energy neutrons generated by reaction [37]. The energies carried by them cannot then be "kept" in the gas by collision, but rather are received directly by the walls of the reaction chamber. This is a direct energy leak and makes the quenching problem very much more difficult. In fact, out of the reaction [37], only the kinetic energy of $^3\text{He}^3$ is kept within the gaseous mixture. This energy is only $1/4$ of the total given by [40], or 1.31×10^{-6} ergs. The energy produced by the reaction [36] is of course retained in the gaseous mixture and is equal to the difference of energies given by [39] and [40] or 1.22×10^{-6} ergs. Hence 50 per cent of the deuteron-deuteron reactions have an effective energy production of only

$$(1.31 + 1.22) \times 10^{-6} = 2.53 \times 10^{-6} \text{ ergs} \dots\dots\dots [50]$$

The average of the reaction energy kept in the mixture is thus, using [39]

$$\frac{1}{2} [2.53 + 6.46] \times 10^{-6} \text{ ergs} = 4.50 \times 10^{-6} \text{ ergs} \dots\dots\dots [51]$$

Compared with gross energy production given by [44], this is only 69.6 per cent; 30.4 per cent of energy produced is delivered directly to the solid walls of the chamber.

Out of the energy kept in the reacting mixture, given by [51] for one single binary reaction or

$$1.606 \times 10^{10} \text{ cal/gr} \dots\dots\dots [52]$$

a good fraction will be absorbed by the reacting deuterium in entering the flame: The deuterium gas is heated, dissociated, and finally ionized to reach the full flame temperature of say 1.600×10^8 K. According to Sanger (1), to heat up to this temperature, the deuterium takes up approximately

$$10^9 \text{ cal/gr} = 0.1 \times 10^{10} \text{ cal/gr} \dots\dots\dots [53]$$

Now the crucial question is: how many grams of deuterium have to be heated in order that one gram of deuterium will be burned to completion? In other words, what is the combustion efficiency of the flame in the reaction chamber? By comparing [52] with [53], it is seen that if 16.06 grams of deuterium have to be heated to flame temperature to get one gram of deuterium burned, then there will be no heat left to be conducted and radiated to the wall *through* the gaseous mixture. But there must be heat conducted and radiated to the wall because the wall, being necessarily of solid material, must be at a temperature, say 2000 K, which is very much lower than the flame temperature of 1.600×10^8 K. This shows that the ratio of mass to be heated and actually burned must be less than 16.06.

For lack of more accurate information, consider a combustion efficiency of the flame zone to be $1/6$. That is, six grams of deuterium have to be heated to have one gram actually burned. Then the energy available for conduction and radiation to the wall per gram of deuterium burned is, according to [52] and [53]

$$(1.606 - 6 \times 0.1) \times 10^{10} \text{ cal/gr} = 1.006 \times 10^{10} \text{ cal/gr} \dots\dots [54]$$

Therefore, by comparing with [45], only less than one-half of the gross energy production is available for "cooling" loss. In fact, with [46], the "cooling" loss energy Q_c produced per unit vol of flame per unit time is

$$Q_c = \frac{1.006}{2.31} \times 0.874 \text{ cal/cm}^3 \text{ sec} = 0.382 \text{ cal/cm}^3 \text{ sec} \dots\dots [55]$$

Now let it be assumed that the flame in the 100 m diam reaction chamber be a cylindrical volume of some 60 m diam and 120 m long. Then within this 120 m of flame, the wall will receive by conduction and radiation *through* the mixture, a heat flux density q_c equal to

$$q_c = \frac{\pi}{4} \times 6000^2 \times 0.382 \\ \pi \times 10,000 = 343 \text{ cal/cm}^2 \text{ sec} \\ = 8.75 \text{ Btu/in.}^2 \text{ sec} \dots\dots\dots [56]$$

This corresponds to a black body radiation at 3990 K.

The question is, of course, whether the heat flux q_c actually equals that given by [56]. For the specified conditions in the reaction chamber, if the actual q_c is larger than [56], then the critical reaction chamber diameter must be larger than the assumed 100 m. If less, then the critical diameter can be smaller. Therefore, one of the basic problems of thermonuclear reaction chamber design is the calculation of q_c or radiation heat flux through a gas layer of variable composition and variable temperature. The technical complication here is, of course, the fact that here the radiation mean free path is large in comparison with the physical dimensions and therefore the simple method developed by astrophysicists for the interior of stars is not applicable. On the other hand, all essential basic information required for the calculation is now available. The problem is thus only complicated but not insurmountable. But, in any event, the flame is almost transparent due to the low density and almost complete ionization. In fact, within the flame, radiation

will come almost only from the specially introduced heavy atoms which are, however, of very, very low density. Therefore, the flame, although of extremely high temperature, is a relatively weak radiator. Hence the comparatively low effective black body temperature of 3990 K may not be far from being correct.

5 Thermonuclear Power Station

The part of energy directly transmitted to the wall by fast neutrons is the difference between [45] and [52]; thus the total energy flux to the wall q is

$$q = \frac{\pi}{4} \times 6000^2 \times \left[0.382 + 0.874 \times \frac{2.31 - 1.606}{2.31} \right] = \pi \times 10,000$$

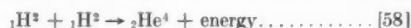
$$582 \text{ cal/cm}^2 \text{ sec} = 14.89 \text{ Btu/in.}^2 \text{ sec.} \dots [57]$$

Although nearly one-half of this energy is realized by slowing down the fast neutrons and is thus distributed in a layer of wall material, not merely delivered at the surface of the wall, nevertheless the tremendous heat density poses a cooling problem which cannot be solved by conventional cooling methods. It seems that the only feasible method is that of transpiration cooling. That is, the wall is made of porous material, say porous carbon or graphite, and cold deuterium gas is forced by pressure through the wall into the reaction chamber. Heat in the wall is picked up by the coolant gas and returned to the reaction chamber. By using a large enough quantity of coolant gas, the wall temperature can be kept at the desired low temperature of, say, 2000 K. In fact, the application of transpiration cooling to nuclear reactors has already been considered by Kaeppler (7). He, however, has not included the important "spacing heating" effects of neutron slowing-down.

Behind the section of reaction chamber occupied by the flame zone, the heat flux due to neutrons is greatly reduced; then the coolant gas forced into the reaction chamber merely serves to lower the temperature of the gas from the flame zone (exhaust gas). At the end of the reaction chamber, the temperature across the chamber cross section should be fairly uniform and at, say, 1000 K. The discharge pressure of this body of hot gas is of course essentially the chamber pressure which is taken to be 100 atm in the above discussion. The high pressure hot gas can be used to generate power through a gas turbine. It is perhaps worth while to note that the product gas is expected to contain only the weakly radioactive ${}^3\text{H}$ and thus should give no difficulty for the power generating machinery. The exhaust from the turbine after being cooled by heat exchanger will pass through the waste extrac-

tion system for removing the nuclear "ash." The purified gas will contain of course mainly D_2 , but also has a very small concentration of H_2 and T_2 produced by reaction [36] and [41]. This gas together with small amount of make-up D_2 to replace the deuterium burned is then compressed to high pressure and fed through the porous wall back to the reaction chamber. This then completes the cycle of the power plant. Fig. 1 is a diagram representing the components and the process of the system.

Of course, by cycling the gaseous mixture repeatedly through the reaction chamber, the concentration of H_2 and T_2 will build up and eventually participate importantly in the energy production through reaction [42], and other reactions. These reactions actually produce the nuclear ash ${}^3\text{He}^4$. Finally the composition of feed gas to the reaction chamber will be stabilized at a fixed ratio of H_2 , D_2 , and T_2 with the production of H and T by reactions [36] and [41] balanced by consumption of H and T through reactions [42] and others. Then the over-all result is that of feeding in deuterium D_2 and taking out ${}^3\text{He}^4$. Thus the reaction chamber effectively converts deuterons to helium according to



and the energy produced per gram of deuterium burned is very much larger than given by [45]. Furthermore, in this final stage the reaction scheme is considerably more complicated than that discussed in Section 3 and the volume rate of energy generation and heat flux must be somewhat different from that calculated previously. However, these calculations will not be attempted here since the purpose of this study is merely to give a general outline of the problem.

6 Ignition

According to the studies made by Sanger (1), the ignition temperature, i.e., the temperature at which the rate of energy production is just balanced by the rate of energy lost, mainly through radiation, with the deuteron-deuteron reaction is approximately 10^7 K. Naturally the question is how can the thermonuclear reaction be initiated by heating the gaseous mixture to this very high temperature. Before the advent of nuclear fission and the fission bomb, such high temperatures seemed unapproachable. But now this is definitely not so. It may even be possible to obtain ignition without using the fission reaction. But at the moment, one can only say that ignition of the thermonuclear reaction is certainly possible; no detailed scheme can, however, be suggested.

7 Thermonuclear Power Industry

The rate of energy production Q according to [46] is 0.00365 kw/cm² in the flame zone. If, as assumed previously, the flame zone is a cylinder of 60 m diam and 120 m long, the total energy production is

$$0.00365 \times (\pi/4) \times 6000^2 \times 12,000 = 1.238 \times 10^9 \text{ kw.} [59]$$

If the thermodynamic efficiency of the power plant cycle is 25 per cent, the power of the station is

$$0.25 \times 1.238 \times 10^9 \text{ kw} = 0.309 \times 10^9 \text{ kw.} \dots [60]$$

Thus continuous operation of the plant will product annually electric energy of the amount

$$0.309 \times 10^9 \times 24 \times 365 = 2.71 \times 10^{12} \text{ kw hr.} \dots [61]$$

In 1954, the annual electric energy production in the United States was approximately 0.5×10^{12} kw hr. Thus in one thermonuclear power plant, perhaps one of minimum size, the capacity is over five times the total effective capacity of the United States! This points to the extreme importance of determining the critical quenching size of the thermonuclear reaction chamber accurately. The speculations in the pre-

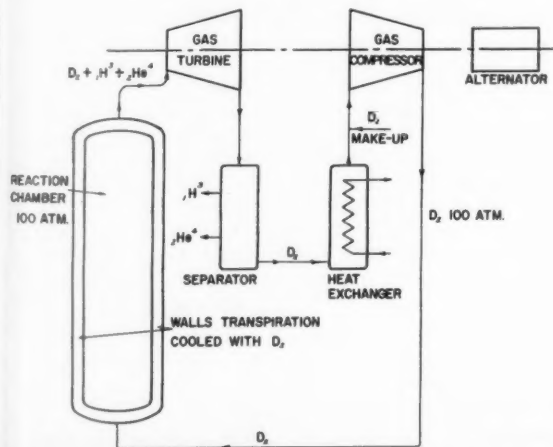


Fig. 1 Schematic diagram of thermonuclear power station

ceding sections are based upon an assumed diameter of 100 m for the reaction chamber. Smaller size and lower flame temperature will naturally reduce the scale of the power plant. However for the deuteron-deuteron reaction, the ignition temperature is roughly 10^7 K. For steady burning, the flame temperature cannot be below this temperature.

To conclude this discussion, comparison of thermonuclear energy and other energy sources will be made: According to [45], the fusion energy of deuterium is

$$2.31 \times 10^{10} \times 1.8 \text{ Btu/lb} = 4.16 \times 10^{10} \text{ Btu/lb of D}_2 \dots [62]$$

The fission of one pound of U-235 gives 3.14×10^{10} Btu. Therefore fusion energy is almost 4/3 times as large as fission energy. Since the natural isotope concentration of deuterium in hydrogen is 1:7000, in terms of natural hydrogen, the fusion energy is

$$(4.16 \times 10^{10})/7000 = 5.94 \times 10^6 \text{ Btu/lb of hydrogen} \dots [63]$$

or, referred to water

$$(5.94 \times 10^6)/9 = 6.60 \times 10^5 \text{ Btu/lb of H}_2\text{O} \dots [64]$$

If the average chemical energy of coal is taken as 11,000 Btu/lb, one pound of water is potentially equivalent to sixty pounds of coal! But even all this is based upon only partial burning of deuteron to triton and proton. With complete burning into ${}^3\text{He}^4$, the thermonuclear energy of deuterium will be still larger. Therefore, if thermonuclear power plants can actually be constructed, then the source of fusion energy far exceeds the other terrestrial energy resources, chemical or fission.

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Comments on Hsue-shen Tsien Paper¹

JESSE L. GREENSTEIN²

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DR. TSIENT had little contact with physicists or astronomers during his studies of a thermonuclear power plant, so that the methods used in his derivation of reaction yields are not the most modern. The concept of effective collision diameter [9, 11] is far from precise; however, as pointed out following [22] it is necessary to fit the experimental data empirically, so that essentially only the forms of [16, 22] are needed. This fitting, or even a different integration method,

(Continued on page 576)

¹ Prof. Greenstein had been asked by the editors to review the Tsien paper. His report recommended that it be published, but he expressed reservations about some aspects of the analysis. Because of Dr. Tsien's absence from the country, the points at issue could not be resolved in the usual manner by private correspondence. Accordingly, Prof. Greenstein gave us his permission to publish his comments, and it is left to the reader to form his own conclusions.

² Professor, California Institute of Technology.

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Climatization of Animal Capsules During Upper Stratosphere Balloon Flights

DAVID G. SIMONS¹ and DRUEY P. PARKS²

Holloman Air Development Center, Holloman Air Force Base, N. Mex.

Experiments requiring exposure of animals to primary cosmic radiation at altitudes above 90,000 ft on 24-hr balloon flights have led to the development of environmental control techniques. Simple methods for supplying oxygen and removing carbon dioxide were developed. Internal capsule temperatures have been maintained at very nearly room temperature on 24-hr flights by providing adequate insulation to use animal heat at night, and by cooling the capsule with a water-can cooler during the daytime.

Introduction

THE frontier of manned controlled flight is inexorably ascending, penetrating one-by-one, atmospheric conditions which are biologically equivalent to space. The life-supporting function of atmospheric pressure is equivalent to space above 63,000 ft. The protecting blanket of air which prevents penetration of primary cosmic particles below 75,000 ft becomes space equivalent in this respect above the 120,000-ft region. When the nature of primary cosmic particles in the upper stratosphere was first established about five years ago, Schaefer (1)³ and Tobias (2) calculated the biological hazard they might pose. All but a few of the slowest and lightest nuclei constituting primary cosmic radiation are too heavy and energetic to be duplicated by even the most modern nuclear accelerators. For instance, they cannot produce one billion electron volt (BEV)/nucleon iron nuclei which are an example of the minimum energy heavy particles that are considered most hazardous biologically.

The known engineering problems challenging flight in the upper stratosphere are formidable. It is of paramount importance that questions such as the necessity and practicability of shielding from cosmic ray primaries be resolved, especially if it can be proved that no serious hazard to manned flight exists for at least limited periods of time. During the past four years, the Aero Medical Field Laboratory of Holloman Air Development Center, N. Mex., has been conducting balloon-borne biological experiments to determine experimentally the degree and nature of this hazard.

The animals selected for the experiments: hamsters, mice, guinea pigs, and monkeys, require more exacting environmental control than men. They cannot pull on a jacket if it becomes uncomfortably cold, nor can they adjust controls to suit their needs. The animal capsules were designed to maintain a minimum pressure of 13 to 14 psi, and temperatures between 65 and 80 F. Carbon dioxide absorbing apparatus was used, but no special consideration was given to humidity control. It was not found necessary.

The design of the animal capsule (3, 4) was determined by equipment available, and by the requirement for a minimum mass of absorbing material above the specimens to permit

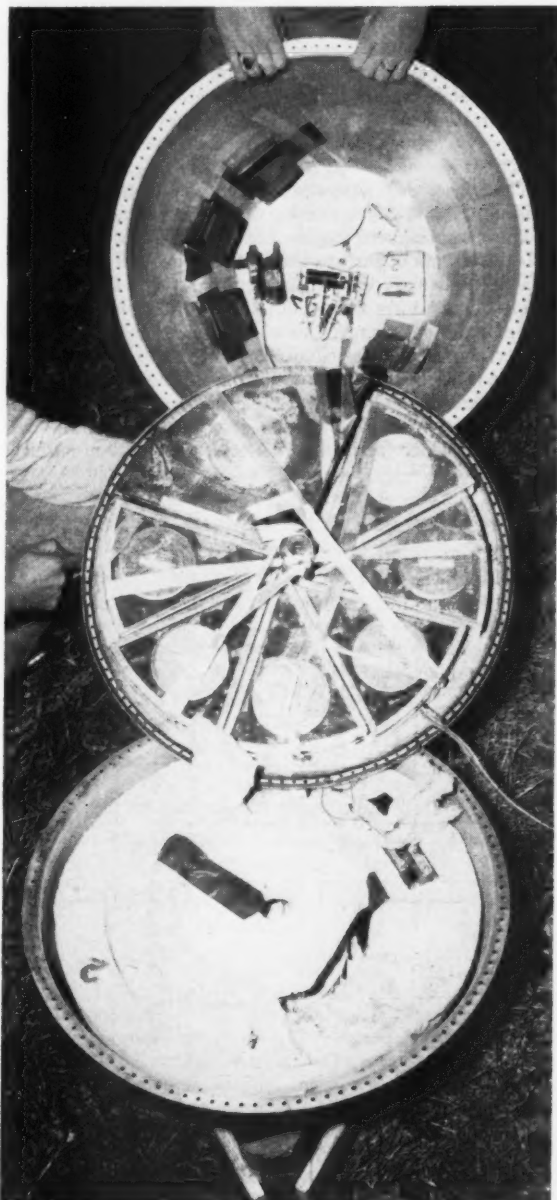


Fig. 1 View of AMFL capsule loaded for flight showing from above down: aluminum cover containing the temperature and pressure coder and recorder, oxygen demand valve, and boxes of experimental specimens; load plate carrying wedge-shaped cages loaded with black mice; and aluminum bottom section containing tube of soda lime, batteries, and water-can cooler in the center

Presented at the Annual Meeting ARS, Chicago, Ill., Nov. 14-18, 1955.

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³ Numbers in parentheses indicate References at end of paper.

maximum exposure to cosmic particles. To obtain the most exposure possible, each flight was usually designed for 24- to 30-hr duration. Balloon performance (5) is strongly dependent upon payload weight; therefore, each control technique and every change in design was critically evaluated to obtain maximum reliability with minimum weight.

The animal capsule illustrated in Fig. 1 consists of two $1\frac{1}{32}$ -in.-thick hemispheres of spun aluminum which seal between them a load plate that carries the experimental animals. The 24-in. diam pressurized sphere is enclosed in a 2-in.-thick cover of styrafoam or lockfoam to provide thermal insulation. Oxygen cylinders are attached to the load harness which forms a sling that carries the capsule on the balloon load line.

Temperature Control

Temperature control considerations are influenced profoundly by the fact that heat equilibrium at troposphere pressures is established largely by air conduction whereas, in the upper stratosphere, it is determined almost exclusively by radiation. Conductive heat exchange between the capsule and the air above 100,000 ft has proved negligible. At the earth's surface, air temperature and rate of air flow are very significant factors in determining capsule temperature, but in the upper stratosphere they make little difference.

Measurements of surface temperatures on a capsule floating at altitude showed that during the daytime the top and bottom temperatures were about 130 and 30 F, respectively, while at night they averaged about -100 and -50 F, (6). The number and altitude of clouds influenced the bottom temperature considerably. Clouds during the daytime increased the temperature, and clouds at night decreased it. At night, the altitude of the clouds made a marked difference; the higher and colder the clouds, the colder the bottom surface

of the capsule. Thus, temperature control for a daytime 8-hr flight, poses an entirely different problem than control for a continuous 24-hr flight, since the same system must then handle both the day and night situations.

The temperature record obtained from AMFL Flight No. 61 (Fig. 2) is an excellent example of a day flight of an insulated, aluminum-surfaced sphere floating in the upper stratosphere without any internal heat source. The capsule was chilled during its ascent through the -60 F heat conductive air of the tropopause between 40,000 and 60,000 ft. At floating altitude of approximately 130,000 ft, the capsule surface radiation equilibrium temperatures averaged about room temperature—proved by its stable internal temperature of 68 F. At night such a capsule would become insufferably cold, and it would become insufferably hot during the day if more than a few heat-producing animals were placed in it.

Experience proved that three inches of styrafoam or lockfoam insulation was sufficient to keep a capsule full of mice or guinea pigs comfortable at night without any additional heat source. With a method for removing the heat produced by the animals during the daytime, excellent temperature control could be expected throughout a 24-hr flight.

A refrigeration system was devised based on the fact that water boils at least ten degrees below room temperature at the low atmospheric pressure found above 90,000 ft. In fact, theoretically at 112,000 ft it boils at 32 F. The can is simply vented to the outside so that it will remain cold at altitude. The system used is illustrated in Fig. 3. The cooler water can is placed in an insulated chamber below the animals. Whenever the animal chamber above becomes too warm, a thermostat activates a blower which circulates the warm air around the cooler-can. Since each gram of water that boils away takes nearly 600 calories with it, this has proved to be a very efficient cooling system.

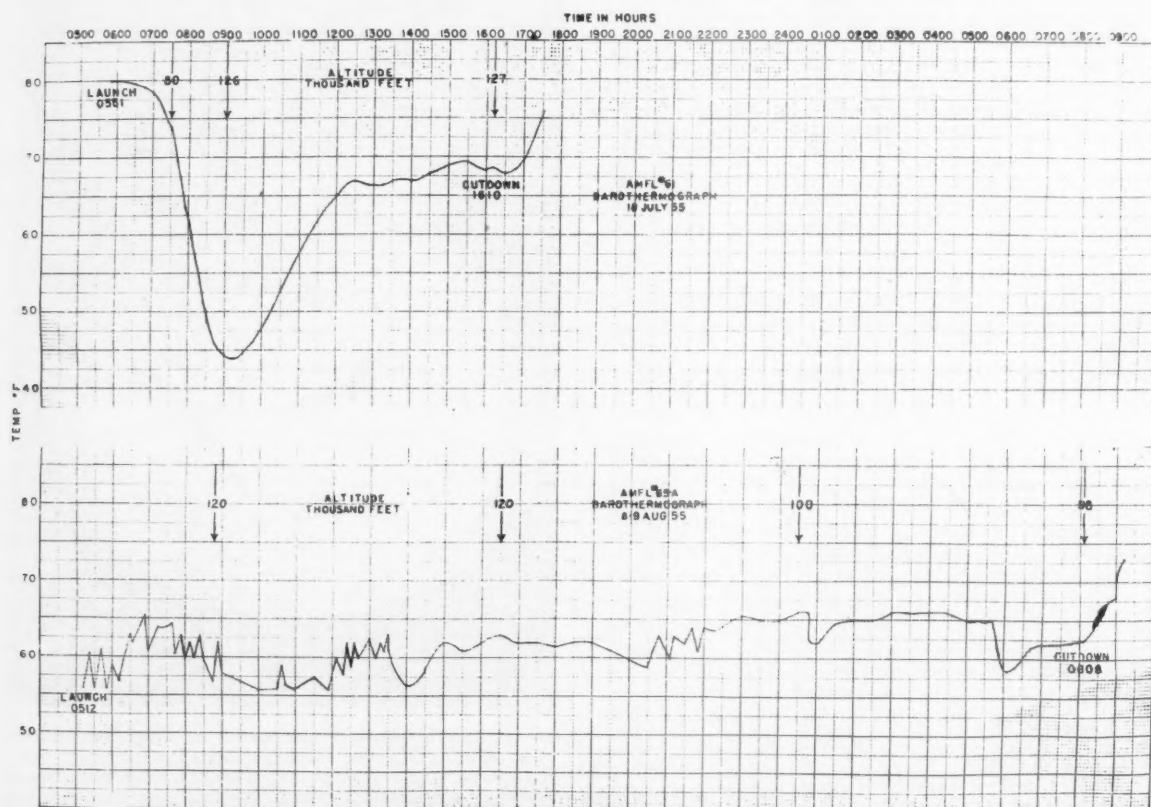


Fig. 2 Records of internal capsule temperature during a daylight flight without internal heat source or cooling (flight #61) and during a 24-hr flight with animals and the thermostatically controlled cooling system (flight #65)

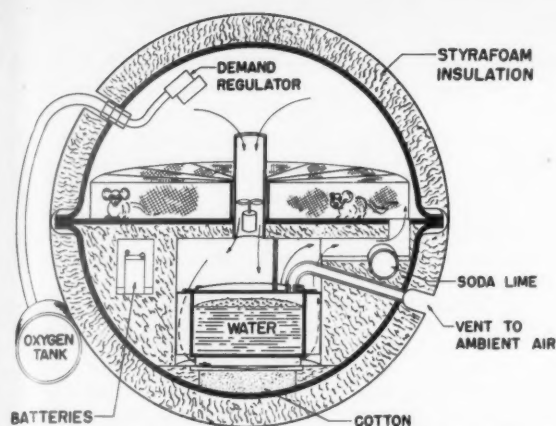


Fig. 3 Schematic of AMFL animal capsule showing the oxygen supply and water-can cooling systems

The temperature record of AMFL Flight No. 65 (Fig. 2) shows the control obtained on a flight carrying 12 guinea pigs above 95,000 ft for 24 hr. The temperature oscillations occurring approximately every 15 min are the result of varying length of duty cycles of the cooling fan which was operated by a hypersensitive thermostwitch. The cooling fan was on continuously from 1345 to 2030. The variation in heat load between the morning and afternoon is apparently due to variations in cloudiness beneath the balloon. The morning was clear, the afternoon broken cloudy. The cooling fan operated only once during the night at 0045, in spite of the sensitive thermostwitch and warm recorded temperature. Most likely it did not operate more frequently because the thermostwitch was much more closely coupled, thermally, to the aluminum capsule wall than the recording element. The wall temperature is considerably below capsule air temperature at night since it is the primary means of heat loss from the capsule. During the daytime the upper half of the capsule wall represents a heat source since the outer surface temperature is 30 or more deg warmer than the capsule air temperature.

Carbon Dioxide Absorption

Approximately 85 per cent of the volume of oxygen consumed by mammals is exhaled as carbon dioxide. The maximum tolerable level within the capsule is about 3 per cent. Thus it can be seen that a continuously operating reliable system for removal of carbon dioxide is essential. The commonest and most readily available absorbent is soda lime. Deliquescent materials such as potassium hydroxide were considered unsatisfactory because of difficulty controlling the highly corrosive liquid produced. Granular lithium hydroxide, which has been used in other applications, would offer a saving in weight, although it is considerably more expensive.

The absorbent material was simply placed in screened containers on early flights carrying relatively few animals. With increasing animal loads, it became necessary to use a forced draft system to insure efficient utilization of the absorbent. A simple fan blower forced air through a polyethylene tube filled with soda lime. This technique has proved very satisfactory and reliable.

Humidity

The necessity to control humidity, from a purely medical point of view, depends directly on temperature. At low room temperatures, a high relative humidity causes little discomfort. When ambient temperature approaches body temperature (above 95 F) a high relative humidity becomes critical.

Under flight conditions the capsule humidity dropped during the warm daytime, and increased only at night when the capsule was cool, making increased humidity tolerable. By carefully minimizing factors that increased humidity, and taking advantage of factors which decreased it, no special humidity absorbing device was required.

The sources of water vapor within the capsule were expired air and urine, the latter being much the larger source. Therefore, urine was held by an absorbent pad placed between the cages and the nonporous load plate, permitting evaporation upward only.

Water vapor was precipitated on the cooler can, capsule walls, and absorbed by the carbon dioxide absorber. Operation of the water-can cooling system during the daytime maintained the relative humidity between 60 and 80 per cent. This occurred because the warm air from the animal chamber (see Fig. 3) was circulated around the water can which was 20 to 40 deg cooler. Condensed water was drained to an absorbent pad capturing it as a liquid to prevent its return to vapor. At night the heat produced by the animals was lost entirely through the capsule walls. As this situation reached equilibrium, the capsule walls became 5 to 10 deg cooler than the air in the capsule. Water condensing on these walls held the relative humidity between 90 and 95 per cent. The soda lime carbon dioxide absorbing unit was adjusted to a low rate of air flow, both to minimize battery drain and to prevent saturation of the absorbent with water vapor.

By contrast, the situation in the capsule after landing on the ground is critical, both for heat and humidity control. The water-can cooler is inoperative below 80,000 ft. The warm air and sunshine on the surface of the capsule add heat so that the capsule walls are warm, not cold, permitting the relative humidity to approach 100 per cent at the same time the temperature is quickly rising.

Oxygen Supply

Arbitrarily selecting a minimum tolerable oxygen partial pressure of 80 mm of Hg and considering the vapor tension of water, minimum capsule pressures for atmospheres of various oxygen percentages are presented in Table 1. The minimum pressure indicates the approximate pressure which must remain in the capsule to prevent the animals from experiencing hypoxia.

Table 1 O₂% and pressure of capsule atmosphere providing sea level alveolar oxygen partial pressure

Oxygen %	Capsule pressure psi
21	14.7
40	8.3
50	6.7
80	4.4
100	3.6

Thus, an intracapsular atmosphere of 100 per cent oxygen at 15 psi provides oxygen for the animals for an additional four hours. However, a pure oxygen atmosphere poses a very serious fire hazard, and most mammals develop pulmonary congestion when subjected to more than 80 per cent oxygen atmosphere at 15 psi for more than 12 to 24 hr. A 50 per cent oxygen atmosphere has been selected as a fair compromise among the various factors.

It is interesting to note that in the closed system employed on these flights the percentage of oxygen should not change, if a constant pressure is maintained from a pure oxygen source. Oxygen within the capsule is metabolized by the animals to carbon dioxide and water. The gaseous component, CO₂, is then absorbed completely by soda lime resulting in a pressure deficit due entirely to loss of oxygen which is replaced by pure oxygen. The nitrogen is neither

metabolized, nor released in significant amounts. The percentage of atmospheric oxygen increases only if there is a leak in the capsule permitting the escape of nitrogen which is replaced by oxygen.

Numerous methods of supplying oxygen have been considered. First, the capsule was simply flushed to 100 per cent oxygen and pressurized slightly. This technique permits a few animals to be encapsulated for a day or two, or a full load of animals for five hours. Whenever the internal capsule pressure has dropped to less than about 8 psi by the time of descent into denser atmosphere, the capsule has been crushed by external pressurization. Standard Air Force oxygen cylinders using both 1800-psi and 450-psi working pressures were considered. The high pressure cylinders represented no weight saving and introduced a filling problem. The greater bulk of the 450-lb cylinders was no problem when they were mounted outside the capsule. Liquid oxygen saved very little weight because of the small quantity required, would be more complicated to handle, and presented a logistic problem. Fig. 3 shows a diagram of the oxygen system used. Standard Air Force D-2 and G-2 cylinders were used, depending upon the volume of oxygen required. The cylinders were connected to each other and to the capsule by flexible hose so that the oxygen lines would not be broken at the time of landing. A series of failures was experienced using copper tubing. The pressure demand valve in the capsule was a two-stage unit which maintained capsule pressure within one psi even though cylinder pressure varied from 35 to 450 psi.

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(Continued from page 554)

than can be obtained in the conventional shock tube. This possibility is of special interest for materials testing because it could be applied to the development of a short duration blowdown tunnel with high stagnation temperature conditions. The flow entering the tunnel would remain relatively uniform for much longer periods of time than could be obtained in the conventional hypersonic shock tunnel. It is possible, using this tailored method, to obtain testing times per foot of shock tube six times as long as can be obtained in

the previously described conventional shock tunnel (8). A program is currently being undertaken at Cornell Aeronautical Laboratory to investigate the possibility of using a shock tube of this nature for materials testing purposes.

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Calculation of a Mollier Diagram for the Decomposition Products of Aqueous Hydrogen Peroxide Solutions of 90 Weight Per Cent H_2O_2 Content

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An enthalpy-entropy (Mollier) diagram was designed for the products of decomposition of 90 weight per cent aqueous H_2O_2 solution. The method of calculation is described. Metric technical units were used. The pressure range covered is 0.1 to 100 kg/cm². Reference state is 0 C temperature, one kg/cm² pressure, liquid water, and gaseous oxygen.

Nomenclature

- a = weight fraction of a component
- ata = technical atmosphere = 1 kg/cm² absolute
- h = enthalpy, kcal/kg, reference state: $t = 0^\circ \text{C}$, $p = 0$, liquid H_2O
- \ln = natural logarithm
- \log = decadic logarithm
- M = molecular weight, kg/mol
- p = partial pressure, ata = kg/cm² absolute
- P = total pressure, $P = p_{\text{H}_2\text{O}} + p_{\text{O}_2}$, sum of partial pressures
- s = entropy, kcal/kg-deg C, reference state: $t = 0^\circ \text{C}$ and $P = 1$ ata, liquid H_2O
- t = temperature, $^\circ\text{C}$
- T = temperature, $^\circ\text{K}$
- v = specific volume, m³/kg
- x = mole fraction of a component

Subscripts

- g = gaseous
- H_2O = water, gaseous, as product of decomposition
- O_2 = oxygen as product of decomposition
- PO = hydrogen peroxide solutions where PO-90 indicates 90 weight per cent H_2O_2
- sat = saturation
- w = water, liquid

Introduction

MOLLIER diagrams for decomposition products from H_2O_2 solutions of 70, 75, and 80 weight per cent concentration are available from (1)² and (2).

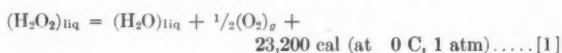
More recently, manufacturers have been able to produce higher concentrations. Ninety weight per cent strength has become a well-established standard item on which much research and development has been based. It was, therefore, thought worth while to calculate and design an enthalpy-entropy (Mollier) diagram for this particular strength.

To facilitate comparison with the above quoted Mollier diagrams for lower concentrations, the metric system and the reference temperature of 0 C were retained.

The procedure of calculation is described.

Weight Fraction (a) and Mol Fraction (x) of Decomposition Products

The equation of decomposition of pure H_2O_2 is



PO-90 decomposes into the weight fractions

$$\begin{aligned} a_{\text{H}_2\text{O}} &= 0.900 (0.5296) + 0.100 = 0.5766 \\ a_{\text{O}_2} &= 0.900 (0.4704) = 0.4234 \\ &1.0000 \end{aligned}$$

The mol fractions are

$$\begin{aligned} x_{\text{H}_2\text{O}} &= 0.7075 \\ x_{\text{O}_2} &= 0.2925 \\ &1.0000 \end{aligned}$$

Assuming Dalton's Law to hold accurately, which is the case only with perfect gases, the fractional partial pressure is equal to the mol fraction, and

$$p_{\text{H}_2\text{O}} = x_{\text{H}_2\text{O}} \cdot P \dots [2]$$

$$p_{\text{O}_2} = x_{\text{O}_2} \cdot P = P - p_{\text{H}_2\text{O}} \dots [2a]$$

The pressure unit used here is the "technical atmosphere," defined as one kg per cm². The "physical atmosphere" is then

$$(\text{phys}) \text{ atm} = 760 \text{ Torr} = 1.033 \text{ ata (kg/cm}^2\text{)}$$

Enthalpy in Superheated Region

The enthalpy of the gas mixture is the sum of the partial enthalpies. The enthalpy unit is kcal/kg.

The partial enthalpies refer to the temperature of the mixture, and the partial pressure of the component.

The data of steam were taken from (3). Since the "Dampf-tabellen" go up to 550 C only, data for higher temperature were taken from (4) and converted into metric units.

The data for oxygen were taken from (5). Oxygen is considered a perfect gas and, therefore, its enthalpy independent of pressure. In the range of the diagram, the largest error thus introduced is only 0.25 per cent of the enthalpy for PO-100 products at 20 ata and 200 C and even smaller for lower H_2O_2 concentrations.

The enthalpy of the mixture is then

$$h = a_{\text{H}_2\text{O}} \cdot h_{\text{H}_2\text{O}} + a_{\text{O}_2} \cdot h_{\text{O}_2} \dots [3]$$

Enthalpy at Saturation

At a given total pressure, the saturation temperature of steam is found from the table of saturated steam as the temperature corresponding to the partial pressure of steam.

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² Numbers in parentheses indicate References at end of paper.

The enthalpy h_{H_2O} at this temperature and partial pressure is interpolated on the same table, and from (5), we interpolate h_{O_2} for t_{sat} .

Enthalpy in Supersaturated Region (Wet Steam Region)

The weight fraction of liquid water is introduced as an additional independent variable designated a_w . The liquefaction of part of the steam changes the composition of the gas phase and, therewith, the partial pressures.

$$\begin{aligned} a_{H_2O} &= \overline{a_{H_2O}} - a_w \\ a_{O_2} &= 1 - a_{H_2O} \end{aligned}$$

where $\overline{a_{H_2O}}$ designates total water content and a_{H_2O} water content in gas phase only.

The mol fraction for the gas phase is calculated from a_{H_2O} and a_{O_2} , excluding a_w , to yield the partial pressure of steam, p_{H_2O} . The saturation temperature for this p_{H_2O} is found from steam table. The corresponding enthalpies are interpolated and the total enthalpy is

$$h = a_w(h_w) + a_{H_2O}(h_{H_2O}) + a_{O_2}(h_{O_2}) \dots \dots \dots [4]$$

If the change of partial pressure (and saturation temperature) with liquefaction is neglected, the result is about 0.8 per cent higher, at the most, than the more accurate result above. Because of this small difference, the *isotherms* are omitted on the diagram in the supersaturated region, as they very nearly coincide with isobars.

Entropy

The entropy of the mixture of decomposition products is composed of the partial entropies of the components.

The partial entropy of steam is found from the steam tables for the partial pressure of steam.

Oxygen is considered as a perfect gas and its entropy is composed of the standard entropy as found in tables for 1.0 ata and the temperature t , and of a pressure term

$$\begin{aligned} s_{O_2} &= s(t, 1 \text{ ata}) - R/32 \log_e (p_{O_2}/1.033) = \\ &= s(t, 1 \text{ ata}) - 0.1430 \log_{10} p_{O_2} + 0.0020 \dots [5] \end{aligned}$$

The corrective term, +0.0020, becomes necessary because the standard entropies in literature are commonly based on $p = 1 \text{ atm} = 760 \text{ Torr}$ and not on $1 \text{ ata} = 1 \text{ kg/cm}^2$. Neglecting this difference would introduce an error of about 0.3–0.1 per cent.

Entropy in Superheated Region

$$\begin{aligned} s_{t,p} &= a_{H_2O} \cdot s_{H_2O}(t, p_{H_2O}) + \\ &+ a_{O_2} [s_{O_2}(t, 1 \text{ ata}) - 0.1430 \log_{10} p_{O_2} + 0.0020] \dots [6] \end{aligned}$$

To interpolate entropy between pressures and temperatures, we must remember that the absolute molar entropy of a perfect gas

$$S = C_p \ln T - R \ln P \dots \dots \dots [7]$$

and the commonly used entropy difference with respect to a reference level of T_0 and P_0

$$S - S_0 = C_p \ln (T/T_0) - R \ln (P/P_0) \dots \dots \dots [7a]$$

Therefore, the entropy change with pressure at constant temperature is proportional to the logarithm of reciprocal pressure

$$\frac{s_{p_1} - s_{p_2}}{s_{p_1} - s_{p_2}} = \frac{\log p/p_1}{\log p_2/p_1} \dots \dots \dots [8]$$

At constant pressure, we interpolate, if s_{T_1} and s_{T_2} are given

$$\frac{s_{T_2} - s_T}{s_{T_1} - s_{T_1}} = \frac{\log (T_2/T)}{\log (T_2/T_1)} \dots \dots \dots [9]$$

where we neglect the change of C_p over the range $T_2 - T_1$.

The error incurred by linear interpolation is only about 0.5 per cent.

To avoid some need for log calculation, a graph of s_{O_2} (1 ata) versus $\log T$ was used.

Entropy at Saturation

Proceed analogously as before with enthalpy and entropy in superheated region.

Entropy in Supersaturated Region

The entropy of the liquid water is found in tables for saturated steam. The procedure is analogous to the one with enthalpy.

The entropy of the mixture is composed of

$$s = a_w(s_w) + a_{H_2O}(s_{H_2O}) + a_{O_2}(s_{O_2}) \dots \dots \dots [10]$$

Specific Volume

The unit of the specific volume is m^3/kg . The density of the mixture is the sum of the densities of the components at their partial pressure and the specific volume is the reciprocal of the density

$$v = \frac{1}{1/v_{H_2O} + 1/v_{O_2}} \dots \dots \dots [11]$$

A general procedure to derive the specific volume is as follows: The specific volume of steam is picked from the steam tables for temperature, t , and partial pressure, p_{H_2O} . The specific volume of water which is needed in the supersaturated region is likewise found in the steam tables. The specific volume of oxygen is easily calculated from the equation of state for a perfect gas with a satisfactory accuracy.

To interpolate v_{H_2O} or v in a large pressure interval, we replace the obvious procedure $v_p = (v_1 p_1/p)$ or $= (v_2 p_2/p)$ by the more accurate procedure of first interpolating the pv product linearly for p and dividing by the pressure p .

$$v_p = \left[p_1 v_1 - (p_1 v_1 - p_2 v_2) \frac{p - p_1}{p_2 - p_1} \right] \frac{1}{p} \dots \dots \dots [12]$$

For O_2 , we calculate

$$v_{O_2} = \frac{26.496 \cdot T}{p_{O_2}} \dots \dots \dots [13]$$

where

$$R/M = \frac{847.87}{32} = 26.496 \left[\frac{m}{^\circ K} \right]$$

Derivation of Isochores

For deriving the desired isochores of 5.0, 3.0, 2.0, etc., m^3/kg , it is most convenient to find the entropy coordinates along an isotherm.

Given are the entropy coordinates for two pressures, p_1 and p_2 , at constant temperature t ; s_1 and s_2 . Let the intersection of the isochore at $v = \text{const}$ with the isotherm at $t = \text{const}$ be the desired point.

As isothermal entropy differences, $s_1 - s_2$, are proportional to the log of the pressure ratio P_2/P_1 ; they will conversely be proportional to $\log v_1/v_2$. With real gases, this is an approximation only, and the intervals should be taken small.

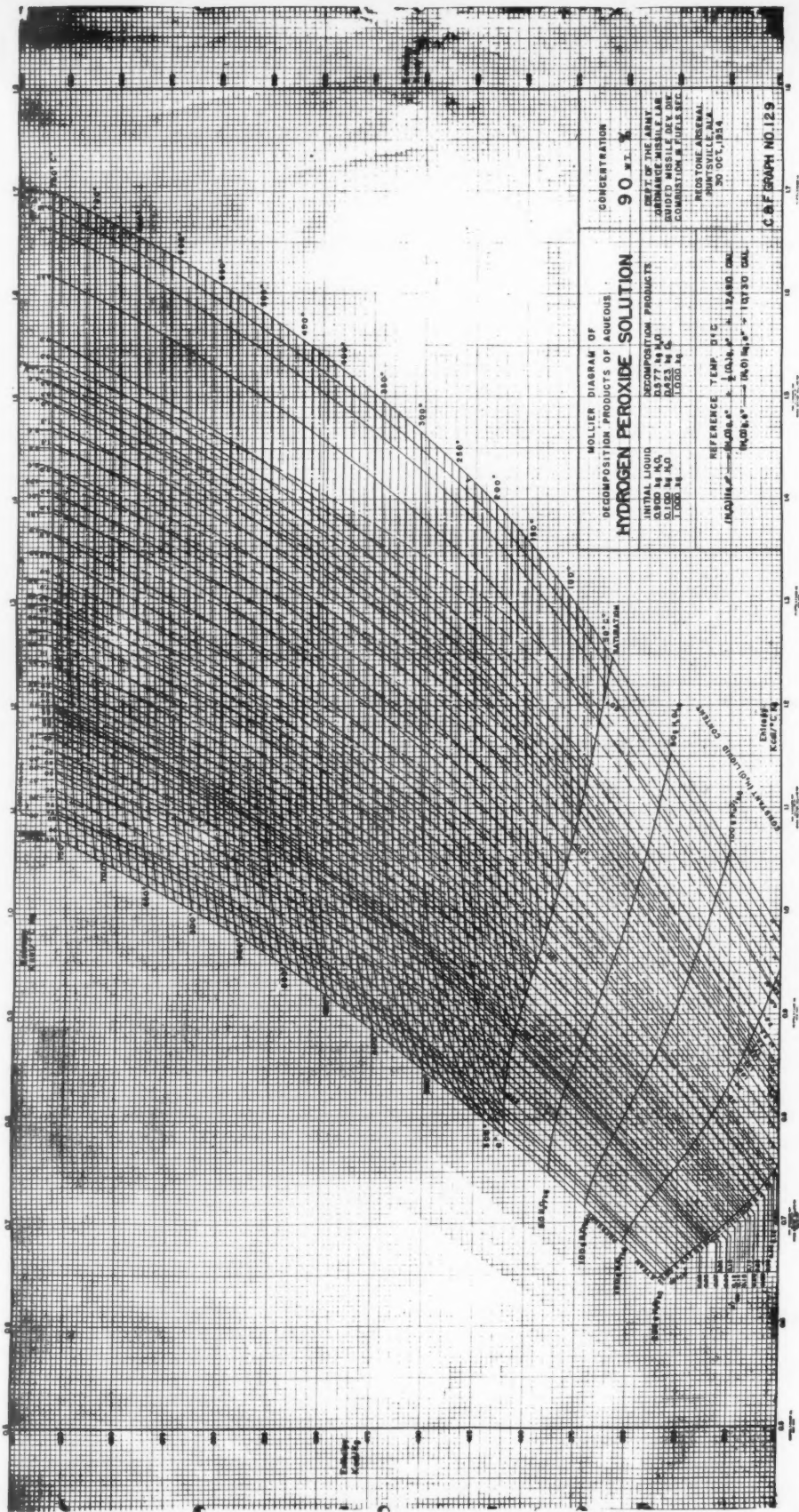
$$\frac{s_1 - s}{s_1 - s_2} = \frac{\log v_1/v}{\log v_1/v_2} \dots \dots \dots [14]$$

Design of the Diagram

Data of enthalpy and entropy as functions of temperature (in steps of 50 C) and pressure (0.1, 1, 5, 10, 25, 50, 75, 100 ata) were tabulated.

The tables of enthalpy versus temperature and pressure,

(Continued on page 576)



Blueprints of the diagram can be obtained from the author in sizes 12 × 22 in. or 30 × 50 in.

Rapid Estimation of Specific Impulse of Solid Propellants

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The short method of Hirschfelder and Sherman for the calculation of the thermodynamic properties of gun propellants is adapted to the quick estimation of specific impulse of solid propellants for rockets. When this approximate procedure is applied to a series of propellants with flame temperatures equal to or greater than 2000 K and specific impulses of 200 to 245 lbf sec lbf⁻¹, at 1000 psia the results are within about 2 per cent of those given by the detailed thermodynamic method which assumes mobile equilibrium. For a propellant ingredient which forms only gaseous products, it is necessary to know only the empirical formula and heat of formation from which a set of three additive constants is calculated in less than ten minutes. The specific impulse of any particular combination can then be estimated from the additive constants for the components in less than five minutes. The method is limited to cases with negligible dissociation and to fuel-rich mixtures.

Introduction

PREPARATORY to the formulation of new solid propellants or the synthesis of new propellant ingredients, it is convenient to be able to estimate rapidly the specific impulse of the propellant or the contribution to specific impulse made by the new material so that the less promising ones can be eliminated without resort to time-consuming experimental work. In the present paper simple equations are derived and applied to the quick estimation of specific impulse of solid propellants. The method used in deriving these equations is similar to that used by Hirschfelder and Sherman (1a)² in the approximate calculation of the thermochemical properties of gun propellants.³ It is assumed that burnt propellant gas consists only of nitrogen, hydrogen chloride, and those components involved in the water-gas reaction; and that the composition of this gas is frozen at an average value determined by the elementary composition together with a more or less arbitrary assumption. The heat capacity and heat content of this gas can be estimated with these assumptions; and from these values the specific impulse is estimated. In the following sections general equations are derived for carrying out these operations, an example is given, and a comparison is made with results obtained by detailed calculations.

Derivation of Equations

The equation for specific impulse

$$I = \sqrt{0.434 nRT_c \left(\frac{2\gamma}{\gamma-1} \right) \left[1 - \left(\frac{p_e}{p_c} \right)^{(\gamma-1)/\gamma} \right]} \dots [1]$$

is transformed into the more convenient form

$$I_{sp} = 0.932 \sqrt{C_p T_c \left[1 - \left(\frac{p_e}{p_c} \right)^{nR/C_p} \right]} \dots [2]$$

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¹ Associate Director, Propellant Development Dept. Mem. ARS.

² Numbers in parentheses indicate References at end of paper.

³ For a detailed explanation of how the Hirschfelder-Sherman method can be used, see (1b).

by use of the following substitutions

$$\gamma = \frac{C_p}{C_v} \quad \gamma - 1 = \frac{nR}{C_v} \dots [3, 4]$$

where

- I_{sp} = the specific impulse in lb force sec lb mass⁻¹
- n = the number of mols of gas produced per 100 grams of propellant
- R = the gas constant, 1.987 cal per mol per degree Kelvin
- T_c = the isobaric adiabatic flame temp in degrees Kelvin
- C_p = the heat capacity of the gases at constant pressure in cal per 100 grams per degree Kelvin
- C_v = the heat capacity at constant volume
- γ = the ratio of C_p to C_v
- p_e = the exit pressure of the gases
- p_c = the chamber pressure

Because the gases from the propellants which will usually be dealt with are cooled from about 2500 K to about 1000 K, the average heat capacity between these two temperatures is taken as the definition for C_p .

By use of an approximation similar to that suggested by Hirschfelder and Sherman for isochoric conditions, the isobaric adiabatic flame temperature is assumed to be given approximately by

$$T_c = 2500 + \frac{H_{rel}}{C_p^{3000K-2000K}} \dots [5]$$

in which H_{rel} is the heat in calories released by 100 grams of the propellant gases in cooling from the flame temperature to 2500 K, and $C_p^{3000K-2000K}$ is the average heat capacity in calories per 100 grams per degree of the gases between 3000 K and 2000 K. Examination of the data on the heat capacity of propellant gas mixtures at various temperatures showed that the average heat capacity between these two temperatures was about 4 to 11 per cent, or on the average about 9 per cent, greater than the average heat capacity for the same gas mixture between 2500 K and 1000 K. Consequently Equation [2] can be written

$$I_{sp} = 0.932 \sqrt{\left[2500 C_p + 0.92 H_{rel} \right] \left[1 - \left(\frac{p_e}{p_c} \right)^{nR/C_p} \right]} \dots [6]$$

We may now define potential P as the function

$$P = \frac{2500 C_p + 0.92 H_{rel}}{100} \dots [7]$$

which when substituted in Equation [6] yields

$$I_{sp} = 9.32 \sqrt{P \left[1 - \left(\frac{p_e}{p_c} \right)^{nR/C_p} \right]} \dots [8]$$

The quantities n , C_p , and P are assumed to be additive functions of the composition of the propellant such that their values may be calculated from the corresponding individual values for the components by use of equations of the type

$$n = \sum_i n_i x_i \dots [9]$$

where n_i is the number of mols of gas produced by the i -th

JET PROPULSION

component, and x_i is the weight fraction of the i -th component in the propellant.

The quantity n_i is estimated from the equation

$$n_i = C + (1/2)(H + N) + HCl \dots \dots \dots [10]$$

in which C , H , and N represent, respectively, the number of gram atoms of carbon, hydrogen, and nitrogen in 100 grams of component, and HCl the number of mols of hydrogen chloride which would be produced in 100 grams of propellant component. If hydrogen chloride is formed, the hydrogen associated with it is not counted in the hydrogen represented by the symbol H in Equation [10].

In deriving the equation for the estimate of C_p , calculated equilibrium compositions of the gases from several resin-oxidizer propellants at 2500 K were examined, and it was noted that the ratio of the number of water molecules to the number of carbon dioxide molecules was between 0.2 and 7. Because many of the values were in the neighborhood of 2 or 3, the value 2.0 was taken as representative. Accordingly, the average heat capacity of a propellant gas between 1000 K and 2500 K is assumed to be given approximately by

$$C_p = 0.67 C_1 + 0.33 C_2 \dots \dots \dots [11]$$

where C_1 is the heat capacity calculated from the elementary composition on the assumption that no carbon dioxide is present, and C_2 is the heat capacity calculated on the assumption that no water is present. C_{pi} is assumed to be given by a similar expression so that

$$C_{pi} = 0.67 [n_{CO}C_{CO} + n_{H_2O}C_{H_2O} + n_{H_2}C_{H_2} + n_{N_2}C_{N_2} + HClC_{HCl}]_1 + 0.33 [n_{CO}C_{CO} + n_{CO_2}C_{CO_2} + n_{H_2}C_{H_2} + n_{N_2}C_{N_2} + HClC_{HCl}]_2 \dots [12]$$

in which n_{CO} , n_{H_2O} , n_{H_2} , n_{N_2} , n_{CO_2} , and HCl are the number of mols of carbon monoxide, water, hydrogen, nitrogen, carbon dioxide, and hydrogen chloride in 100 grams of gas; the symbols such as C_{CO} denote the heat capacity in calories per mol per degree for the particular gas between 1000 K and 2500 K, and the numerals 1 and 2 after the brackets indicate the following two methods by which the values for the gas composition are to be derived from the elementary composition.

Method 1	Method 2
(no carbon dioxide)	(no water)
$n_{CO} = C$	$n_{CO} = 2(C) - O$
$n_{H_2O} = O - C$	$n_{CO_2} = O - C$
$n_{H_2} = \frac{H}{2} - O + C$	$n_{H_2} = \frac{H}{2}$
$HCl = HCl$	$HCl = HCl$
$n_{N_2} = N/2$	$n_{N_2} = N/2$

The resulting expression for C_{pi} is

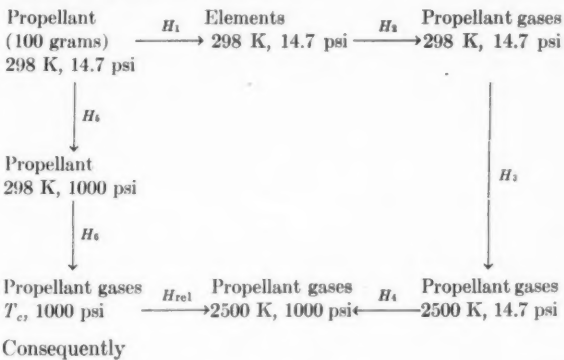
$$C_{pi} = 4.27(C) + 3.98(H) + 4.23(O) + 4.21(N) + 8.27(HCl) \dots [13]$$

The heat capacities used were obtained from the compilation of the National Advisory Committee for Aeronautics (2). This equation is limited to substances which contain only the elements whose symbols appear in the equation. It is a fair approximation of the heat capacity even when the ratio of the concentration of water to that of carbon dioxide is not close to the value of 2 assumed in the derivation of this approximate equation. Because both water and carbon dioxide have high heat capacities compared to those of hydrogen and carbon monoxide, the heat capacity of a mixture of the gases which partake in the water-gas equilibrium is not sensitive to the ratio of concentration of water to carbon dioxide.

The derivation of an expression for P_i involves first the derivation of an equation for H_{rel} , the heat released by cooling 100 grams of propellant gas from the adiabatic flame temperature to 2500 K.

In considering the following thermodynamic cycle the

enthalpy changes H_4 and H_5 are neglected on the assumption that the materials are ideal, and the enthalpy change H_6 is zero because the process is adiabatic



Consequently

$$H_{rel} = H_1 + H_2 + H_3 \dots \dots \dots [14]$$

The heat evolved in the first step H_1 is simply the heat of formation in calories of 100 grams of the propellant from the elements in their standard states at 298 K; consequently for the i -th constituent we may write

$$H_1 = 1000 \Delta H_i \dots \dots \dots [15]$$

in which ΔH_i is the increase in heat content of the system in kilocalories for the formation of 100 grams of the constituent.

The heat evolved in the second step H_2 is given by

$$-H_2 = n_{CO_2} \Delta H_{CO_2} + n_{CO} \Delta H_{CO} + n_{H_2O} \Delta H_{H_2O} + HCl \Delta H_{HCl} \dots [16]$$

in which the standard heats of formation are in calories per mol.

The heat evolved in the third step H_3 is

$$-H_3 = n_{CO_2} H_{CO_2} + n_{CO} H_{CO} + n_{H_2O} H_{H_2O} + n_{N_2} H_{N_2} + n_{H_2} H_{H_2} + HCl H_{HCl} \dots [17]$$

The symbols such as H_{CO_2} represent the heat required to change a mol of the particular gas from 298 K to 2500 K.

The heat released, H_{rel} , is then calculated from Equation [14] and the equation

$$H_{rel} = 0.67 (H_{rel(1)}) + 0.33 (H_{rel(2)}) \dots \dots \dots [18]$$

The subscripts 1 and 2 denote that the calculation is to be made by use of the gas composition calculated by use of the previously described methods 1 and 2. The resulting equation is

$$H_{rel} = 1000 (\Delta H_i) - 44400(C) - 8420(H) + 52900(O) - 8880(N) + 4630(HCl) \dots [19]$$

The heats of formation used were obtained from the compilation by the Bureau of Standards (3). Combination of Equations [7], [13], and [19] yields the following expression for the potential of the i -th constituent

$$P_i = 9.2(\Delta H_i) - 302(C) + 22.0(H) + 592(O) + 23.6(N) + 249(HCl) \dots [20]$$

The expansion term $1 - (p_e/p_c)^{nR/C_p}$ is easily obtained from the plot of C_p/n vs. the expansion term given in Fig. 1.

In summary, the estimation of the specific impulse of a solid propellant at a chamber pressure of 1000 psia and an exhaust pressure of 14.7 psia involves estimating the number of mols of gas by Equation [10], the heat capacity of the gas by Equation [13], the potential of the propellant by Equation [20], the effect of expansion through the nozzle by Fig. 1, and the specific impulse by Equation [8]. For a monopropellant all these manipulations can be carried out in fifteen minutes or less depending on whether a slide rule or desk calculator is used.

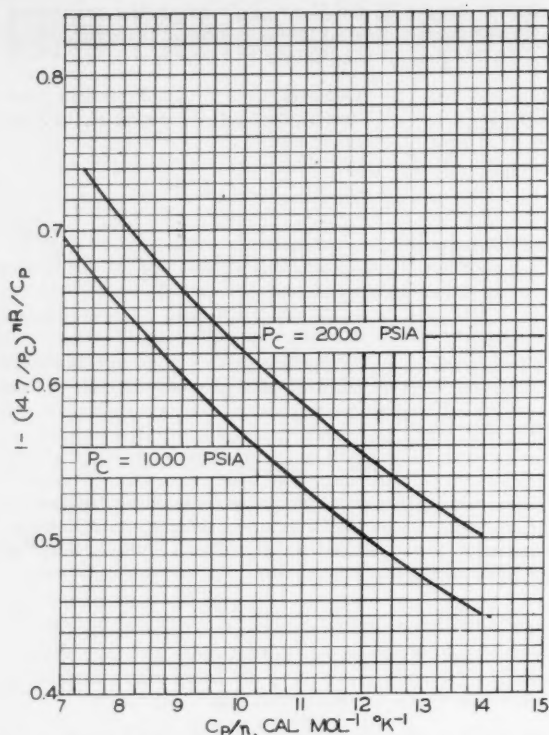


Fig. 1 Expansion term vs. molar heat capacity for use in estimating specific impulse by the short method

Example

An example of the calculation of the required constants for a typical propellant component, nitroglycerin, follows

Formula: $\text{CH}_2\text{NO}_3 \cdot \text{CHNO}_3 \cdot \text{CH}_2\text{NO}_3$
 Formula weight: 227
 Heat of formation: -37 kcal/100 gm
 $C = 1.32$; $H = 2.20$; $O = 3.96$; $N = 1.32$
 $n_i = 3.08 \text{ gm mol/100 gm}$
 $C_{p_i} = 36.7 \text{ cal/100 gm/K}$
 $P_i = 1685 \text{ cal/gm}$

These constants are then tabulated as in Table 1 for future use.

An example of the estimation of the specific impulse of a propellant which contains several components is summarized in Table 2. The example is based on a typical double-base propellant (4) for which the basic constants have already been given in Table 1.

Comparison with Detailed Calculation

Comparisons have been made of the values of specific impulse estimated by this method with those calculated by the detailed thermodynamic method on the assumption that

Table 2 Estimation of specific impulse of a double base propellant

Substance	x_i	$n_i x_i$	$C_{p_i} x_i$	$P_i x_i$
Nitrocellulose (13.25%N)	0.522	2.04	20.3	559
Nitroglycerin	0.435	1.34	16.0	733
Diethyl phthalate	0.033	0.28	1.8	-22
Ethyl centralite	0.010	0.10	0.6	-23
	1.000	3.76	38.7	1247

$C_p/n = 10.3$ and from Fig. 1 the expansion term for a chamber pressure of 1000 psia is 0.558.

$$I_{sp} = 9.32 \sqrt{1247(0.558)} = 246 \text{ lbf sec lbf}^{-1}$$

equilibrium was maintained in the exhaust gas during expansion through the nozzle. The flame temperatures covered the range from 1500 K to 3000 K and the specific impulses a range from 190 to 245 lbf sec lbf⁻¹ at 1000 psia. Generally, the agreement was within 2 per cent and the average error was 1 per cent. In one case for a flame temperature below 2000 K the error was as large as 6 per cent, but this is attributed to the presence of carbon in the propellant gas and the inapplicability of the short method to a system of this type. It is concluded that in new propellant systems the estimates by this quick method are only what their name implies, i.e., estimates. In precise work the short-method estimates must be supplemented by an appropriate detailed calculation to test their accuracy. After this has been accomplished the short method provides a rapid method for the exploration of the magnitude of the effects to be expected in formulation changes.

Comparison of Equations [2] and [8] shows that the isobaric adiabatic flame temperature of the propellant can be estimated by the relationship

$$T_c = \frac{100 P}{C_p} \dots \dots \dots [21]$$

The values obtained by this means are usually within approximately 100 K of the flame temperatures calculated by the detailed method, except in cases for which dissociation is significant or products other than those assumed in the derivation are present in significant amount.

Certain limitations of this short method should be indicated. Care should be taken to use the method only for estimating the specific impulse of propellants containing the stoichiometric quantity or less of oxidizer. Also, the method does not take into account the effects of dissociation at high flame temperatures. This last limitation is not so serious as might be expected, because frequently the specific impulse calculated by the detailed method on the assumption of chemical equilibrium during expansion is not greatly different from that calculated on the assumption of no dissociation and frozen equilibrium. The short method most nearly resembles the latter type of calculation.

Acknowledgments

The author is indebted to Dr. Stanley C. Burket for

Table 1 Additive constants for use in rapid estimation of specific impulse

Substance	ΔH_f kcal/100 gm	Composition, gm atom/100 gm				n_i gm mol/100 gm	C_{p_i} cal/100 gm/K	P_i cal/gm
		C	H	O	N			
Nitrocellulose (13.25%N)	-58 ¹	2.125	2.60	3.66	0.95	3.90	38.9	1071
Nitroglycerin	-37 ²	1.32	2.20	3.96	1.32	3.08	36.7	1685
Diethyl phthalate	-27 ³	5.40	6.30	1.80	...	8.55	55.8	-675
Ethyl centralite	-84 ³	6.35	7.46	0.37	0.75	10.46	61.5	-2290

¹ Ref. (5).
² Ref. (6).
³ Ref. (4) p. 11.

stimulating discussions which formed the basis for part of the analysis presented here, and to Dr. Ralph W. Lawrence for constructive criticism of the derivations of the equations.

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Calculation of a Mollier Diagram

(Continued from page 570)

and of entropy versus temperature and pressure are to be combined into a plot of enthalpy versus entropy with lines of constant temperature (isotherms) and lines of constant pressure (isobars) and lines of constant volume (isochores).

To construct the main isotherms and isobars, the coordinates (h, s) of their intersections were found from the h - t table and the s - t table and plotted. Intermediate temperature steps of 10 C were interpolated linearly along the isobars.

The entropy coordinates for the intermediate isobars were interpolated in logarithmic proportions along isotherms as shown in Equation [8]. In supersaturated region, only isobars are drawn as the isotherms very nearly coincide with them as mentioned previously.

In the superheated region, a closed formula was derived relating the total pressure and the specific volume.

For steam, (6) offers the formula

$$v = \frac{47.06}{P} T - \frac{0.9172}{(T/100)^{2.82}} - P^2 \frac{1.3088 \cdot 10^{-4}}{(T/100)^{14}} + \frac{4.379 \cdot 10^7}{(T/100)^{31.6}} \dots [15]$$

which is valid down to saturation pressure if $t > 300$ C.

For each temperature, we obtain an equation of the form

$$(v_{H_2O})_P = \frac{A}{P} - B - CP^2 [\text{m}^3/\text{kg}] \dots [16]$$

The specific volume of O_2 at the total pressure and at a given temperature is represented

$$(v_{O_2})_P = \frac{26.496 \cdot T}{P \cdot 10^4} = \frac{D}{P} \dots [17]$$

where A, B, C, D are constants at constant temperature. The specific volume of the mixture is composed

$$v = a_{H_2O} (v_{H_2O})_P + a_{O_2} (v_{O_2})_P \dots [18]$$

We substitute $(v_{H_2O})_P$ and $(v_{O_2})_P$ and arrange

$$P^3 + \frac{a_{H_2O} B + v}{a_{H_2O} \cdot C} P - \frac{a_{H_2O} A + a_{O_2} D}{a_{H_2O} \cdot C} = 0 \dots [19]$$

The equation was solved for P in steps of 50 C, inserting chosen values of v . This was done by machine calculation.

The entropy coordinates for these pressures were interpolated between two pressures on the isotherm in logarithmic proportions.

Acknowledgment

The assistance of Ken Brown, John Luckhowec, and Tully Myers in calculating data and designing the diagram is gratefully acknowledged.

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Comments on Hsue-shen Tsien Paper

(Continued from page 564)

is required if a nuclear resonance level exists in the compound nucleus near the energy of the colliding particles. See H. Bethe² for the more complete theory of thermonuclear reactions.

According to [46] the energy production per unit volume is not excessive. In the sun, energy-producing reactions are contained, give only 10^{-4} watts/cm³, and are self-sustaining. But in spite of the high temperatures and moderate energy output in the chamber, I am not convinced that a case is made for the possibility of a thermonuclear furnace without other and more elaborate features for containing the reaction volume to a very small fraction of the chamber volume. In the sun, gas convection and radiation transport the energy through a temperature gradient of 10^{-2} K/cm; in the furnace, the gradient must average 10^4 K/cm. If the gradient of temperature is too steep the reaction stops, and if too slow the walls evaporate. The stellar reaction is automatically self-regulated, since gravitational contraction heats the gas, and containment in a solid chamber is not a problem. The particular feature of Dr. Tsien's calculation that can be directly tested astrophysically is his device for stopping the fast reaction products of [36, 37], $^1H^3$, $^1H^1$, $^2He^3$; the neutrons cannot be prevented from escaping. His suggestion is to introduce a heavy and only partially ionized atom to increase the collision cross section with energetic particles of ionized hydrogen, deuterium, and helium. Intuitively, however, it is difficult to imagine a process, or particle, which moderates other particles by collision and which will not, at high temperatures, produce radiation, and therefore introduce another quenching factor. If the ions were present in sufficient numbers to produce equilibrium between kinetic energy and radiation, the black body temperature would be attained, i.e., 10^8 K; if the integrated total opacity through the gas is t (the optical thickness), the wall temperature $T(R)$ at wall radius R is obtained from the gas temperature $T(r)$, reaction zone radius r , as

$$T(R) = T(r) \cdot (r/R)^{1/2} t^{1/4}$$

Since $T(R)/T(r)$ must be less than 10^{-6} to prevent the walls from evaporating, and since $(r/R)^{1/2}$ cannot be much less than 10^{-1} , the optical thickness t must be less than 10^{-16} , which is extremely low for any real ionized gas. Therefore, the problem mentioned following [56] is a difficult one, as yet unsolved. New technical methods are required for isolating a hot gas volume by material in which a steep temperature gradient exists, and which will not radiate. It should be remembered that at high temperatures radiative transfer usually surpasses convective or conductive transfer.

² Bethe, H., *Physical Review*, vol. 55, 1939, p. 434.

Technical Notes

Continued Investigations of the Opposing Jet Flameholder

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A previous publication by the authors concerning the opposing jet flameholder was preliminary in nature. In this note, selected additional data are presented. Hypotheses concerning the mechanism of stabilization are discussed briefly and investigations now in progress are mentioned.

Introduction

IN RECENT times, studies of flameholding elements other than physical bodies have received considerable attention for several reasons. Theoretically, physical holders such as V-gutters and rods are difficult to analyze, while in practical applications they pose undesirable drag and flame-spreading problems. One different type of flameholder is the opposing gaseous jet, which was described in a preliminary paper (1)³ by the authors. Since the writing of that paper considerable data have been gathered and extended investigations are in progress at the Gas Dynamics Laboratory of Northwestern University.

Before any additional data are presented, the observations in (1) will be reviewed briefly. There the opposing jet was described as a high velocity gaseous stream discharging in a direction 180 deg to the primary flow. A metal tube of about 1/16-in. ID was used for the jet, and supply pressures to the tube were in the vicinity of 100 psig so that the flow was choked. The mass flow rate through the tube was generally less than 1 per cent of the primary mass flow rate. The flame anchored itself about 1 to 3 in. upstream from the end of the metal tube, and flame spreading was satisfactory. Normally, the jet medium was air, although tests with oxygen and nitrogen were also attempted. An opposing air jet stabilized premixed propane-air flames having approach velocities of several hundred fps when the mixture was at atmospheric pressure and was not preheated. Indications were that the performance curve would peak on the rich side. It was observed that a flame could not be stabilized with a pure nitrogen jet. The proposed interpretation was that a critical zone existed in the stagnation region at the upstream tip of the flame and that conditions in this zone determined the stabilization. It was suggested that the performance curve for an air jet might be shifted toward the lean side by making the critical zone richer; that is, by mixing small quantities of fuel with the jet air. Furthermore, it was predicted that higher jet flow rates might result in better flameholder performance.

Selected Additional Data

One purpose of this note is to present some additional data so that a more complete understanding of the properties and potentialities of the opposing jet will be available. Fig. 1 shows flame blowoff data for two different opposing jets; namely, for a pure air jet and for a stoichiometric jet of propane and air. The apparatus and the operating conditions are the same as in (1) except that the jet supply pressure is 70 psig

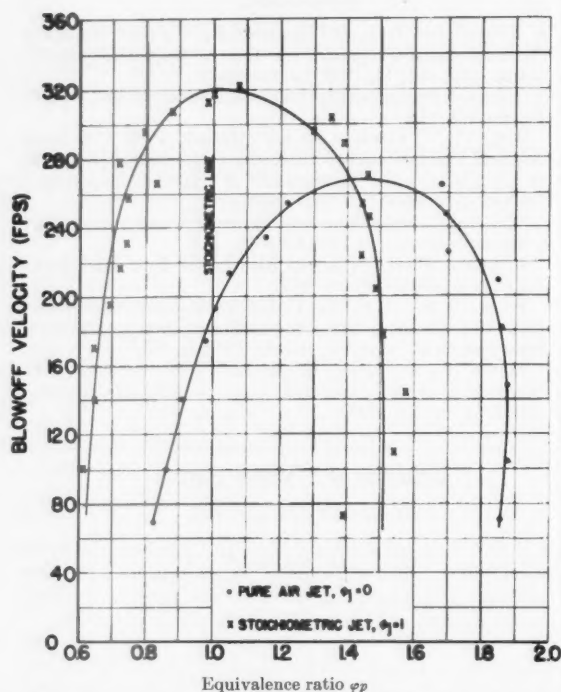


Fig. 1. Blowoff curves for a pure air jet and for a stoichiometric jet. Tube 0.059 in. ID; jet supply pressure 70 psig

instead of 100 psig. Furthermore, the jet tube length is shorter; namely, 4 in. as compared to 18 in. before. The figure shows that as the jet is made richer the blowout curve shifts toward the lean side; that is, toward lower values of the equivalence ratio of the approach stream. Tests with still richer jets, not shown here, have substantiated this observation. The increment of fuel flow necessary to change the jet equivalence ratio by unity is only about 1 per cent of the primary fuel flow. This minute quantity of fuel mixed into the jet causes a relatively large shift in the blowout curve. It is difficult to imagine a small quantity of fuel having such an effect in a premixed stabilized flame unless the fuel feeds into a zone of special importance to the stabilization. Therefore the term critical zone has been employed.

There are several other interesting observations concerning Fig. 1, but these can be discussed to better advantage after a model for the stabilization mechanism has been presented. The model that is being employed follows the concept that a small pilot reactor, or critical zone, exists in the low velocity stagnation region at the nose of the flame. As shown in Fig. 2, there are three streams feeding into the critical zone: the jet media (m_j), the recirculated media that is entrained by the free jet (m_r), and the unburned primary mixture that penetrates directly into the zone from upstream (m_p). Initially the analysis assumes instantaneous mixing of these three streams, and their proportions and compositions determine the equivalence ratio in the critical zone. It seems fair to assume that a criterion for stability is the energy level in the critical zone. The burden for maintaining this energy level falls upon the recirculated media. Thus sufficient burned gases must be entrained by the jet. It has been shown experimentally that recirculation is necessary for the jet to function as a stabilizer. One striking demonstration of the extent of the recirculation patterns is that successful light-off has been

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³ Numbers in parentheses indicate References at end of paper.

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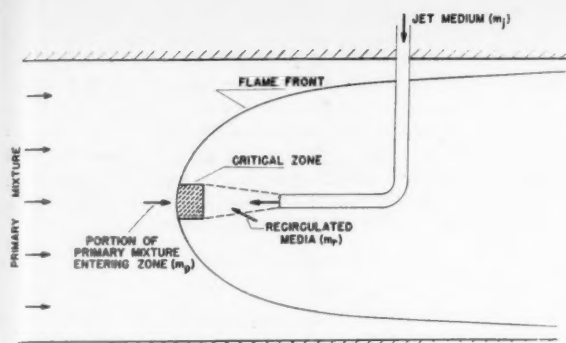


Fig. 2. Model for the stabilization mechanism

accomplished with the spark plug 5 in. downstream from the end of the jet tube.

Returning to Fig. 1, we note that for a stoichiometric jet the performance curve peaks at a primary equivalence ratio of unity. Thus for blowout at the peak of this curve, two of the three streams feeding into the critical zone have equivalence ratios of unity. The third stream is the recirculated media, the composition of which can be assumed to be approximately that of the equilibrium products of combustion for the equivalence ratio of the primary stream. For the condition under consideration, then, the above assumption dictates that the recirculated media be inert. When these three streams are mixed in the critical zone, the result is that the equivalence ratio in the critical zone at the maximum approach velocity is unity. This result could have been predicted a priori by applying to the critical zone knowledge of the functional relationship between reaction rates and equivalence ratio. Furthermore, it seems logical to assume that for any jet mixture strength the equivalence ratio in the critical zone is always unity at the maximum approach velocity. This observation enables us to perform a calculation to determine the relative contributions of the primary stream and the jet stream to the critical zone. In order to perform a simplified calculation it will be necessary to assume that the recirculated media is inert even when the primary mixture is not stoichiometric. This assumption probably introduces only a small error into the calculations. The expression for the equivalence ratio in the critical zone, φ_c , is expressed by

$$\varphi_c = \frac{\sum \text{rates of fuel flow into zone}}{\text{stoichiometric fuel-air ratio in zone} \times \sum \text{rates of air flow into zone}} \quad [1]$$

The inert recirculated media affects the reaction rate in the critical zone but not the equivalence ratio there. Consequently, we can write

$$\varphi_c = \frac{m_p \left(\frac{F_s \varphi_p}{1 + F_s \varphi_p} \right) + m_j \left(\frac{F_s \varphi_j}{1 + F_s \varphi_j} \right)}{\left[m_p \left(\frac{1}{1 + F_s \varphi_p} \right) + m_j \left(\frac{1}{1 + F_s \varphi_j} \right) \right] F_s} \quad [2]$$

where

- φ_c = equivalence ratio in the critical zone
- m_p = mass rate of flow of the primary mixture into the critical zone
- m_j = jet mass flow rate (all of which is assumed to enter the critical zone)
- φ_p = equivalence ratio of the primary mixture
- φ_j = equivalence ratio of the jet
- F_s = stoichiometric fuel-air ratio for the fuel

When we apply Equation [2] to the peak of the curve for the pure air jet, we substitute $\varphi_c = 1$, $\varphi_p = 1.45$, $\varphi_j = 0$, $F_s = 0.0641$, and obtain

$$\frac{m_p}{m_j} = 2.44 \quad [3]$$

Thus the primary stream contributes 2.44 times as much mass to the critical zone as does the jet. It should be pointed out that the computations are valid only for the given jet size at an approach velocity of 270 fps. However, Equation [3] can be used with little error for other jet mixture strengths. If we measure the total mass flow rates of the jet medium and of the primary mixture, we can use Equation [3] to determine the per cent of the primary mixture that enters the critical zone. The value computed in this manner is about 1 1/2 per cent and shows that the critical zone is indeed very small. The preceding computations have also been made with satisfactory results for the data from a jet having an equivalence ratio of 2.

Some information concerning practical applications is available from Fig. 1. Normally afterburners operate with a stoichiometric approach mixture; consequently, it is desirable for the blowout curve of the flameholder to peak at an equivalence ratio of unity. Fig. 1 shows that an opposing jet having itself an equivalence ratio of unity would be required. Therefore, it is probable that investigations as to the practical applications of opposing jets will be concerned principally with stoichiometric opposing jets. Richer jets might be used for light-off; they would also be well suited for modulation when the primary mixture is lean. If, because of its greater simplicity, a pure air jet is employed instead of a stoichiometric jet for an afterburner application in which the approach mixture is stoichiometric, Fig. 1 shows that the maximum approach velocity attainable would be decreased by about 40 per cent.

One aspect of Fig. 1 that is not so easily explained is the fact that the curve for the stoichiometric jet has a higher maximum velocity than does the curve for the pure air jet. An examination of Fig. 3 may provide a partial explanation

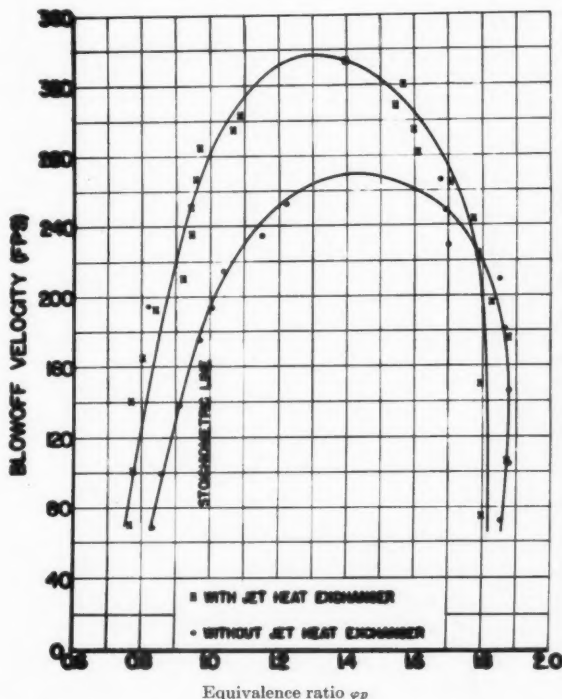


Fig. 3. Blowoff curves showing the effect of a jet heat exchanger. Discharge tube 0.059 in. ID; jet medium air; jet supply pressure 70 psig

for this behavior. Fig. 3 indicates the beneficial effect of preheating the jet medium, which was air for these tests. The lower curve is for the short jet tube utilized in most of the tests recorded in this note. The tube is $\frac{1}{8}$ -in. OD \times 0.059-in. ID \times 4-in. length exposed to flame. The upper curve is for a long tube such as was used in (1). For this latter test the jet air was passed through a $\frac{1}{8}$ -in. tubing heat exchanger in contact with the flame and then was fed to the $\frac{1}{8}$ -in. tube previously described. The figure shows that heating the jet medium aids the stabilization. On the rich side the two curves are drawn as separate lines, but the data scatter is such that they could have been justifiably drawn as a single line. Apparently with rich mixtures a high jet temperature is obtained for the short tube alone and the heat exchanger does little good. The effect of heating the jet medium, while pronounced, is not tremendous because the temperature in the critical zone is influenced primarily by the hot recirculated gases. A comparison of Figs. 1 and 3 suggests that a thermal effect may be responsible for the higher maximum velocity obtained for the stoichiometric jet in Fig. 1. This thermal effect may be the burning that takes place along the boundary of the free jet. The term free jet is defined here as the portion of the jet stream from the end of the tube to the critical zone. Burning is observed to take place along the boundary of the free jet either when the jet medium is a fuel-air mixture or when the medium is pure air and the primary stream is rich. This boundary burning may act like incremental heat addition in aiding the stabilization. Since boundary burning exists at the peak approach velocities for a stoichiometric jet and not for a pure air jet, this burning may explain the higher approach velocities with the stoichiometric jet. However, these higher velocities might also be explained by mixing phenomena in the critical zone. Moreover, beneficial effects of boundary burning may be responsible for the near vertical slope of the rich side of the curve for the pure air jet, although any inference drawn from rich data is open to question because of the inherent rough burning that accompanies rich mixtures.

Fig. 4 is a typical curve taken from the work of Edward

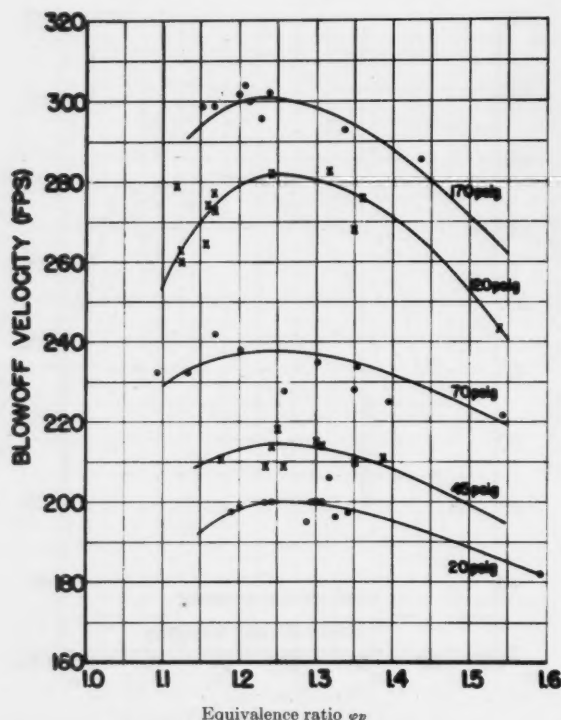


Fig. 4. Tops of blowoff curves at various jet supply pressures for an opposing air jet having a tube 0.033 in. ID

Pohlmann (2), who studied among other things the effects of varying the jet diameter and the mass flow rate. The figure presented here gives a family of performance curves for various supply pressures of pure air to a short jet tube of 0.033-in. ID. Supply pressures to the tube are labeled on the graph. Only the top portions of the curves are given. It is seen that increasing the jet supply pressure, and thus the mass flow rate and momentum, makes possible higher maximum velocities. It can be expected that higher maximum velocities will be accompanied by wider ranges of equivalence ratio.

Acknowledgments

The authors extend their special thanks to Edward Pohlmann for the use of the data depicted in Fig. 4. The studies described in this note were sponsored by the Department of Mechanical Engineering and were made possible by a fellowship grant of the General Motors Corporation as well as by the financial support of Faculty Research Projects 103-54, 103-55, the Technological Institute, Northwestern University.

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- 2 Pohlmann, Edward, "Observations on the Opposing Jet Flame Stabilizer," Internal Report, Gas Dynamics Laboratory, Northwestern University, October 1955.



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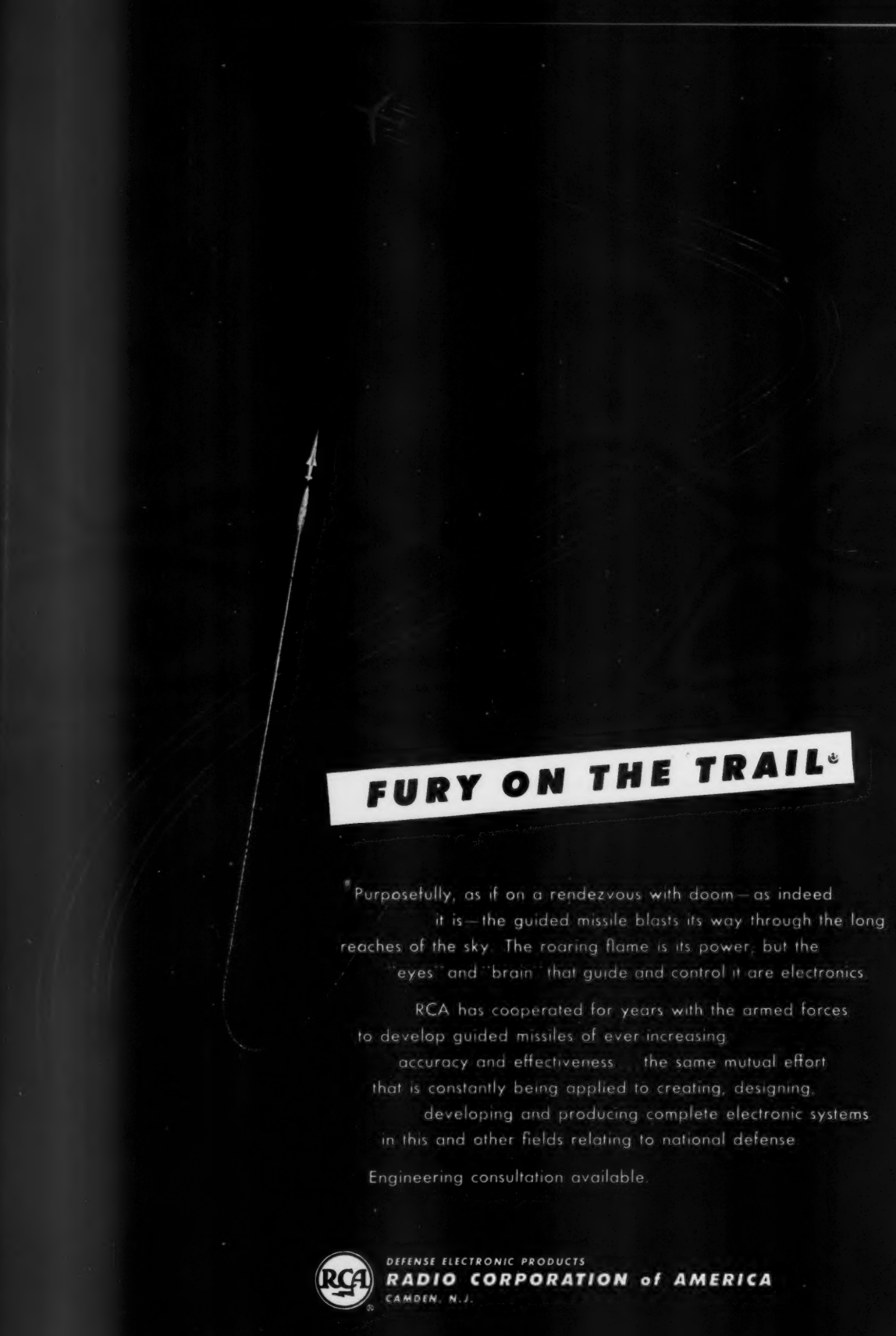
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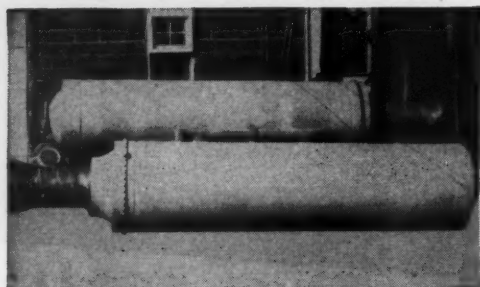


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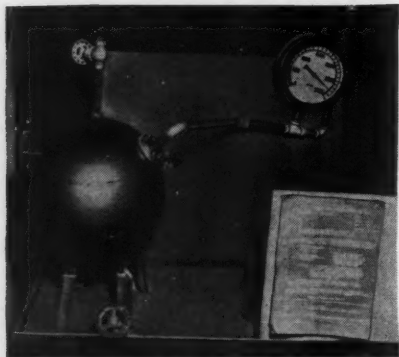
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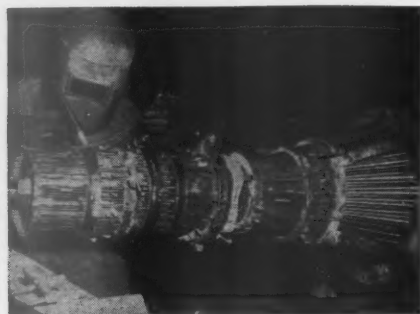
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Jet Propulsion News

Alfred J. Zaehring, American Rocket Company, Associate Editor
Norman L. Baker, Indiana Technical College, Contributor

MORE ON PROJECT VANGUARD

WITH the addition of recently released details, the picture of the Vanguard vehicle is now almost complete. It will be 72 ft long and have a maximum diameter of 45 in. Gross weight will be 11 tons. Second stage will carry all guidance: three-axis gyro reference system, pitch programmer, and velocity sensors. Guidance system will orient rocket to horizontal flight; stability will come from spin along longitudinal axis.

The planned orbit of 300 miles may not be reached because of errors in launching and guidance. The orbit will be elliptical: perigee, 150 miles; apogee, 1500 miles. Some satellites may carry no payload. Launching of the first satellite is slated for early 1958.

Four new subcontractors for Vanguard have been announced. Allegany Ballistics Laboratory (Cumberland, Md.) and Grand Central Rocket Co. (Redlands, Calif.) will design and develop a new solid propellant rocket for the third stage. Presumably the ABL model will use double-base propellant while Grand Central will furnish a composite variety. The Minneapolis-Honeywell Regulator Co. (Minneapolis, Minn.) will design and build a three-axis reference system which consists of gyros that indicate roll, pitch, and yaw. Vickers Electric (St. Louis, Mo.) will design and make the magnetic amplifier autopilot which will control the flight. The autopilot will take its instructions from the gyro system and will keep the vehicle on course by moving the gimbal-mounted rocket engine.

Naval Research Laboratory has developed the "Minitrack" radio angle tracking system for the satellite. The satellite will have a miniature transmitter (108 mc) with an output of 10-50 mw operating on silver-zinc batteries. Satellite signal will be received and tracked by ground stations. Data from ground stations will be fed to a computing facility to determine the exact orbit. Minitrack, weighing about 2-3 lb, may also telemeter data from satellite. Ground turn-on will allow telemeter transmission when the satellite is over a ground station.

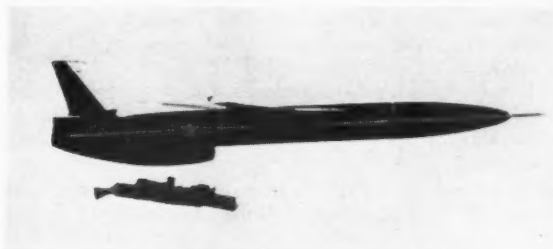
BALLISTIC MISSILES

- Titan, the second USAF intercontinental missile, will be developed by Martin at Denver. It will be a two-stage rocket.
- Atlas will be produced at Sorrento (Calif.) by Convair. Facilities contracts for the missile were awarded as follows: AVCO (Stratford, Conn.), \$1,425,000; Burroughs Corp. (Detroit), \$400,000; General Electric (Syracuse), \$861,000; Sperry-Rand (St. Paul), \$155,000; Western Electric (New York), \$500,000; North American Aviation (Los Angeles), \$6,463,000; Air Products (Allentown, Pa.), \$5,768,000.

Burroughs Corp., developing a computer system for the missile, reports that the Atlas will be guided while under power. The missile will be tracked by radar and the tracking data fed into a computer. The computer will compare this data with correct trajectory, make corrections in missile's flight path whenever it deviates from the established course.

● Thor, a single-stage IRBM, is being developed by Douglas Aircraft for USAF. Subcontractors are General Electric (warhead), North American Aviation (rocket engine), and AC Sparkplug (guidance).

● Jupiter, a ship-launched rocket, is the new Chrysler-Army-Navy IRBM. Development will begin under a \$15 million allocation by the Navy for 1957, pending an additional \$72 million contract for actual production.



SNARK SM-62: Speed is a big questionmark

- Activation of the first unit to fire the Redstone missile was announced by the Secretary of the Army. Formed at Redstone Arsenal, the unit will be designated the 217th Field Artillery Missile Battalion (Redstone).

CRUISE MISSILES

- Status of Snark (photo) is still undecided. Although the Northrop SM-62 has intercontinental range and could carry a nuclear warhead, it is a subsonic (about Mach 0.9) cruise missile. Other specifications: length about 60 ft; span about 50 ft; diameter about 5 ft. The missile recently made a 2000-mile test flight at Patrick AFB (Fla.)
- Navaho SM-104A (formerly SM-64) ramjet, now undergoing flight tests at Patrick AFB (Fla.), will be produced by North American under a \$5,325,000 government contract.
- Launching dollies for carrier launching of the Regulus are jettisoned after each launch according to Chance Vought. The three-wheel cart, made of tubing, weighs $\frac{1}{20}$ of the old launching platform and RATO bottles. In launching, the dolly is hooked to the carrier's steam catapult system. The carts are also used for stowing and handling the missiles aboard.
- The guided-missile submarine SSG will become the nuclear-powered, guided-missile submarine SSGN. Two conventional attack submarines Grayback (SS-574) and Growler (SS-577) now under construction will be completed as guided missile submarines. All three subs will be capable of launching the Regulus.

GROUND-TO-AIR MISSILES

- The U. S. Army is sponsoring the development of an antimissile missile by Cornell Aeronautical Lab.
- Talos surface-to-air missile will be used by the Continental Air Defense Command to protect U.S.A.F. bases. Developed by the Navy and Bendix, Talos will also be used aboard Navy ships.
- Coordination of Nike batteries will be the job of the new Missile Master (MM) System developed by Martin and the Army. MM is a complete electronic system for coordinating and directing a large number of Nike batteries. The system collects information on the identity and location of aircraft and presents this information on electronic displays. The data are then distributed to firing batteries. Each MM is housed in a large two-story building (Antiaircraft Operations Center) where radar data gives an over-all view of the air situation. Tracking operators monitor early warning information; tactical controllers evaluate progress of an engagement and assign targets; friendly protectors insure that Nike batteries do not fire on known friendly aircraft; battery

commanders select or receive target assignments without conflicting with or duplicating the actions of other batteries. MM can operate in conjunction with the Air Force SAGE system and also can be used with future missile systems. Other contractors on MM were Airborne Instruments Lab and the American Machine & Foundry Co.

● To further increase its effectiveness, a new version of the Nike may carry a nuclear warhead. The warhead, says the Army, would enable one missile to destroy an entire attacking air fleet.

Eventually some Nike batteries will be manned by National Guard units when sufficient missiles are available to replace all conventional guns. Newest site for Nike battery: Okinawa.

● Additional Terrier details have been made known: total length, 27 ft; booster length (solid propellant), 14 ft; sustainer rocket (also solid), 13 ft; booster diameter, 8 in. Booster burns for about 4 sec giving a speed of Mach 2. Missile proper carries a proximity fuze and has a top speed of Mach 2.5. Booster carries a rating of "over 200,000 hp." Estimated cost of the Terrier is about \$30,000.

In operation on the USS Boston, acquisition radar tracks 2 targets at a time. Control radar can handle 8 in-flight missiles per minute. A computer handles target tracking and fire control. The Navy reports that Terriers have intercepted a target near 50,000 ft, destroyed a jet drone flying at over 500 knots.

● Tartar missile, also by Convair, is smaller than Terrier but has a similar layout. Tartar is powered by a single-stage rocket.

GENERAL

● Thrusts greater than those of any previously existing single rockets are claimed for Phillips Petroleum Co.'s solid propellant boosters. Solid propellant of German World War II Rheintochter booster produced 143,000-lb thrust for 0.6 sec while current North American liquid propellant rocket engine for Atlas is pegged at about the 125,000-150,000-lb thrust level. Phillips' propellant uses synthetic rubber, carbon black, and ammonium nitrate.

● First U. S. rocket engine that can be throttled up or down at will was revealed by Curtiss-Wright Corp. The unit is used to power the Bell X-2 rocket plane. C-W began development during World War II under the leadership of Dr. Goddard at Caldwell (N. J.) facility which includes five test cells and a hydraulic test lab. Other rockets will be used for ATO and missile propulsion.



NEW TM-61B MATADOR

Test firings of TM-61B have been successful. The new Martin missile carries a larger nose section, a new airborne guidance system. Estimated cost is about \$85,000

● Dart is an Army Ordnance AT missile. About 6 ft long, it is said to be powered by a solid propellant rocket. Control comes from wires which trail out behind the missile.

● Sparrow missile by Sperry has been operating from carrier-based aircraft in Atlantic and Pacific tests. The air-to-air-missile is powered by an Aerojet-General solid propellant rocket. The Sparrow I is now in a "combat ready" status according to the Navy. Built at Bristol (Tenn.) the bird is 12 ft long and weighs 300 lb. Guided by a radar beam from the launching aircraft, Sparrow has a speed of over 1500 mph.

● Reliability of liquid rocket engines may run as high as 95-96% in future systems according to Reaction Motors engineers.

● "Portable foxholes" will be provided by a small rocket motor weighing 3½ lb. Packed in a cardboard cannister, the rocket blasts a hole in the ground.

● Rocket experts are not immune to crystal-gazing predictions. As witness: Isothermal expansion of gases might eventually be used in an atomic reactor for rocket propulsion according to two Northwestern University scientists, Romer and Cambel. This type of gas flow could lead to smaller rocket engines of greater power. The process would require a gas whose molecules vibrate in an infinite number of ways.

According to a prediction of Simon Ramo of Ramo-Woodbridge Corp., a substantial portion of the country's air freight could be carried by guided missiles within 10 years.

Still further into the future, Kraft Ehrlicke of Convair stated that within 200 years the moon would be colonized by rocket flights from the earth.

FOREIGN ROCKETS

England: Fairey Fireflash may be used as armament on the Hawker Hunter. The air-to-air missile was originally scheduled for use on the Vickers Swift. A beam rider missile, it recently scored hits on target drones and will be the first British guided missile delivered to the R.A.F. Another British missile, a ground-to-air type, is going into production. For ground-to-ground use, Britain will use the U. S. Corporal missile. The ramjet Napier test missile was described by British sources as having reached an altitude greater than any similar British missile. The missile measures 17-20 ft long and uses cluster-type solid propellant booster rockets.

Ready for launching during the IGY will be Great Britain's new high altitude solid propellant research rocket. The rocket is 25 ft long, has a diameter of 17 in., and is designed to carry instruments to an altitude of over 100 miles. The rocket is launched from an adjustable tower 81 ft high and weighing 35 tons. Meanwhile, Britain's Ministry of Supply has established an 8000-acre test center near Scotland for development of an ICBM. But, claim the English, they are about 2-4 years behind the U. S. in missile development.

Germany: Daimler-Benz has reported satisfactory tests on its new ramjet engine designed for helicopter rotor blade propulsion. Thrust of the 7-in. diam engine is about 60 lb. Fuel consumption of the motor is about 8.4 lb/min.

A group of German engineers plan to fire a three-stage rocket during the IGY. Expected altitude of the rocket is about 60,000 ft. The group has already fired 20 rockets since 1954. Launching site will be near Cuxhaven on the North Sea.

Sweden: Sweden is placing a supersonic AA missile in operation. Built by the Swedish Air Force, the Mach 3 missile is two-staged. Sweden is currently spending about a million dollars each for missile development and procurement.

Russia: Soviet Air Force Commander-in-Chief Marshal Pavel Zhigarev hints that top priority is being given to the ICBM (Red designation, SIG). In his policy-making book, "Thoughts on Air Strategy," published last year, Zhigarev

Jet Helicopters Pass in Review



MARINE HRS uses ROR (Rocket on Rotor) to boost lift power, increase safety factor



FLYING IN FORMATION, Hiller HOE-1's go through their paces before joining the Fleet. Powered by two Hiller ramjets, craft cruise at 60 knots



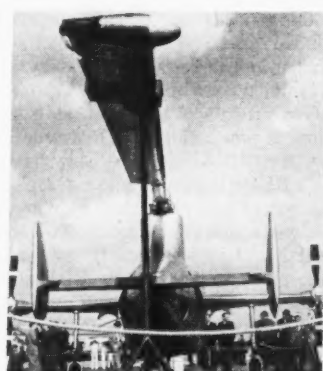
WORLD'S BIGGEST helicopter, Hughes Aircraft turbojet XH-17, can carry 10-ton payload



ONE-MAN HELICOPTERS, Kellett Aircraft's "Stable Mable" (left) and Rotorcraft's "Project Pinwheel" (right) both use RMI peroxide rocket motors



HELICOPTER HYBRID, McDonnell's XV-1, uses conventional pusher props for horizontal flight, pressure jet rotors for vertical flight. Piston engine supplies air for jet units



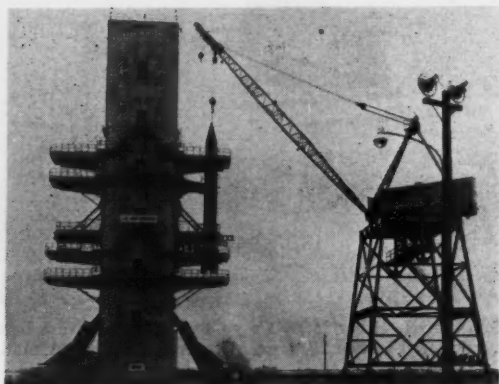
STILL EXPERIMENTAL, Fairy Gyrodome is first British jet helicopter

says that the SIG has made the strategic bomber obsolete. In addition to intercontinental weapons, "space weapons" were mentioned. Meanwhile, Soviet party boss Khrushchev stated that Russia would soon have guided missiles carrying an H-bomb.

Other sources report that Russia has a rocket with a range

of 100-150 miles which can be launched from a submarine 300 ft under the water. The rocket is said to be able to carry a nuclear warhead. The Soviets are said to have a missile launching sub fleet of about 400. Another report says that the Reds are working on a three-stage, manned, intercontinental glider rocket.

Rockets at Redstone

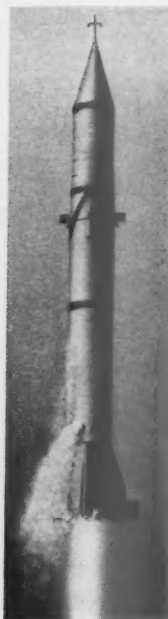


Gantry lowers REDSTONE missile into test stand

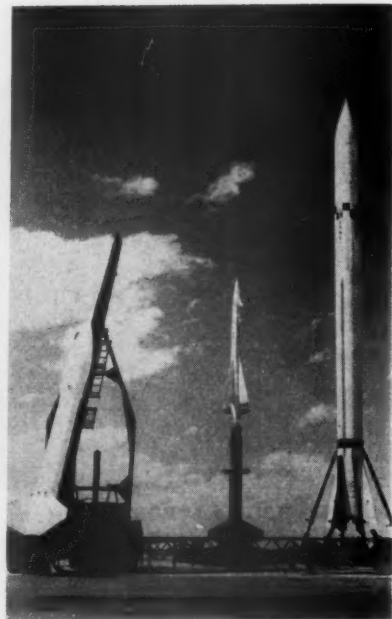
REDSTONE ARSENAL was established in 1941 near Huntsville by the Chemical Warfare Service. Located in northern Alabama, it comprises an area of about 40,000 acres in its 10- by 12-mile plot.

In 1948 Redstone was designated as a center for the Army Ordnance Department research and development activities pertaining to rockets and related items. In 1950 rocket activities moved here from Fort Bliss, Texas, with the formation of the Guided Missile Division. This move was made by 500 military personnel, 130 German rocket scientists, 120 General Electric Company contractor employees, and 120 Civil Service employees. In 1951, a division of Thiokol Corp. moved from Elkton, Md., to Line 1 for work on solid rocket propellants. Lines 3 and 4 were occupied by a group from Rohm & Haas Co. (Philadelphia, Pa.) in 1952, also for work in solid propellants. Improvements during this period included: \$170,000 for a Thiokol administration building; a million dollar Gorges Laboratory for Rohm & Haas; and a \$4 million construction program. The Rohm & Haas program is concerned with shoulder-fired rockets. Another industrial firm to join hands with Redstone was the Chrysler Corp. of Detroit, Mich. In 1952 Chrysler began work on developing the REDSTONE missile. Development work was done at Redstone with production at Warren, Mich., and testing split between Redstone Arsenal (static) and Patrick AFB, Fla. (dynamic). Chrysler recently joined forces with Redstone and the Navy to develop an IRBM in the 1000-1500 mile class. Three other industrial firms are also on the base but these are not connected with arsenal work.

The primary mission of Redstone is to serve as a center for research and development of guided missiles, including components and systems development in the fields of aerodynamics, guidance and control, propulsion, assembly tech-



REDSTONE blasts off



Big Three, .HONEST JOHN, NIKE, CORPORAL are Army Ordnance weapons



High Speed Wind Tunnel is part of Aerophysics Lab.



Rocket flashes through accuracy range

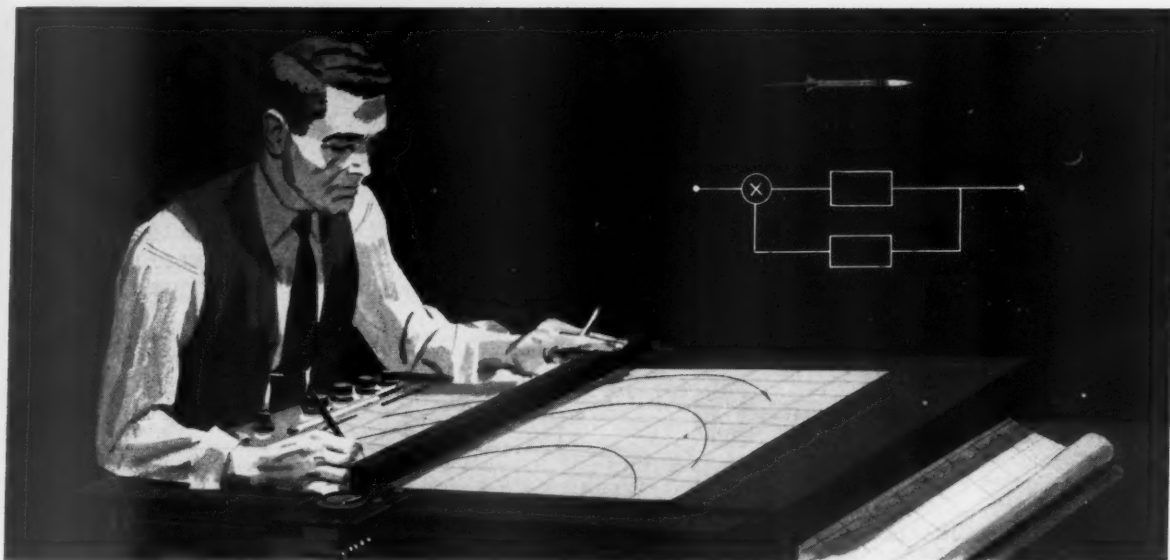
niques, transport, test, and launching. Work is concerned with both liquid and solid propellants. Located here is a giant new 150-ft test stand, a former Peenemuende wind tunnel, and a Guided Missile School.

The Redstone Arsenal payroll for 1954 totaled over \$33 million and arsenal strength included over 7000 military and civilian personnel (excluding contractors such as Thiokol, Rohm & Haas, and Chrysler). In 1954 a \$23 million expansion project was begun. New projects included: a new building for the Ordnance Guided Missile School; a \$3.67 million engineering building; and a \$2 million guidance and control building. With these new developments, rockets may be expected to play an increasing role in American ordnance.

Notable Achievements at JPL

MISSILE GUIDANCE AND CONTROL... In applying advanced servo and noise-theory techniques to missile control systems, JPL has led and advanced the field of missile guidance.

Among specific achievements are the application of Wiener RMS methods to multiple-input, multiple-loop servos, and matching missile trajectory to missile control transfer function for optimum accuracy.



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TELEMETERING

PACKAGING

MECHANICAL ENGINEERING

The Jet Propulsion Laboratory is an organization devoted entirely to scientific research and development. Covering an 80 acre area in the rising foothills of the San Gabriel mountains, north of Pasadena, it occupies an ideal location close to residential districts.

The working staff of the Laboratory consists of about 1250 people, all employed by the California Institute of Technology. The various projects are conducted under continuing contracts with the U. S. Government.

The prime objective of JPL is obtaining basic information in the various sciences related to missile systems development and in all phases of jet propulsion. Underlying the entire Laboratory activity, a major continuous program of fundamental research in the physical sciences is constantly in progress.

In its missile system and jet propulsion undertakings, the Laboratory maintains a broad technical responsibility, from basic research to prototype engineering. By virtue of this and the integrated nature of the JPL technical staff, each individual is drawn into close contact with the general field to which his specialized technical abilities contribute the most.

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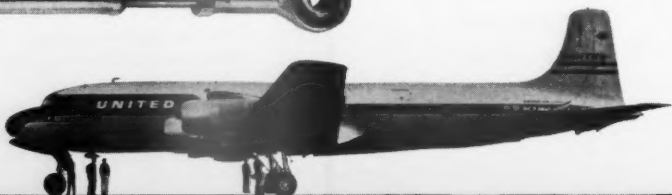
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MISSILES CHIEF Eger V. Murphree, in his first major address since taking public office, tells AMERICAN ROCKET SOCIETY that time is only block to U. S. missile progress

ARS SEMI-ANNUAL MEETING

Cleveland Conclave Draws Over 300

AN estimated 337 engineers, scientists, and students from all over the country assembled at Cleveland's Hotel Statler last month for the June 18-20 Semi-Annual Meeting of the AMERICAN ROCKET SOCIETY held in conjunction with the Semi-Annual Meeting of The American Society of Mechanical Engineers.

The showing of the ARS, by far the younger of the two societies, surprised many of the long-time ARS members who recalled the not-too-distant past when a small conference room and a single banquet table usually sufficed for such meetings.

Host for this meeting was the Cleveland-Akron Section which went all out to make its guests welcome.* Despite hot and humid weather at the beginning and the omnipresent hand of security throughout (responsible for the withdrawal of two papers), the meeting was carried off with a high degree of polish. Here are some of the highlights:

Ramjets: Marcel Piry of Fairchild Engine and Airplane Corp. got the session underway with René Leduc's paper (297-56), "Early Work and Latest Realizations With Ram-Jet Engines." (Dr. Leduc was unable to present his own paper because of recent illness.)

Looking forward to the "near future" when men will fly at Mach 2 without too much difficulty, Dr.

* While many Section members did an outstanding job, special mention should be made of the work done by Paul Ordin of NACA, arrangements chairman; James Daus of NACA, displays; and Henry Burlage, Case Institute, program chairman.

Leduc sees ramjet engines as standard powerplants of future aircraft. Backing this vision, of course, is his long experience—dating back to the early 'thirties—in the ramjet field.

Of even greater significance, perhaps, is the fact that Dr. Leduc has had ramjet aircraft under actual flight tests since 1949. His latest prototype, the 021 now undergoing flight tests near Istres, is still subsonic. But, says Dr. Leduc, it is the last subsonic ramjet he will build. The next will be a truly supersonic airplane.

● Albert Gail of Cornell Aeronautical Lab spoke next on "The Ramprop—A Supersonic Jet-Driven Propeller" (296-56). Essentially a 120-ft-long ramjet, the Ramprop, as envisioned by Mr. Gail, in an upright position would act as a rotor to lift or lower an airplane vertically, then convert to a horizontal position for forward flight. Based on his predicted performance characteristics, he believes the Ramprop would be particularly suitable for aircraft weighing 150,000 lb or more. Use of such a unit, he feels, could, among other things, increase payload capacity 10,000 lb or more, up speed to almost sonic level while matching current fuel economies.

At one time, Office of Naval Research was interested in the Ramprop idea. Now, however, there is practically no interest in the subject, said Mr. Gail.

● Paper 298-56, "Some Fundamental Aspects of Ramjet Propulsion" by Arthur N. Thomas, Jr., of Marquardt Aircraft Co., was a clearance casualty.

Liquid Propellant Rockets: The first paper in the afternoon, "Dynamics of a Pump-Fed, Variable-Thrust, Bi-Propellant, Liquid Rocket Engine System" (299-56), was a joint effort by Marvin R. Gore and John J. Carroll of Aerojet-General Corp. Aimed primarily at those interested in control design, this paper established the principal dynamics of a bi-propellant engine over a broad thrust range by analog computer simulation of the nonlinear physical elements, then went on to analyze the performance of the rocket engine with typical controllers.

● The next paper, "A Solid-Liquid Rocket Propellant System" (301-56), was also a joint effort. Co-authors were George E. Moore and Kurt Berman of General Electric. Dr. Moore presented the paper in which was described a self-igniting hybrid rocket propellant system employing 90% hydrogen peroxide as the oxidizer and polyethylene as the fuel. Dr. Moore reviewed the reasoning leading to interest in such a system and discussed briefly some of the project findings.

● Getting his paper (302-56) through clearance too late to be preprinted, John C. Becker, Jr., of Aerojet-General finished up the first-day sessions with his interesting "Evaluation of Various Liquid Rocket Exhaust Jet Flame Suppressants." In an attempt to lessen the heat of jet flames—and at the same time by-pass complicated, heavy, expensive cooling systems, Mr. Becker tried four chemical additives as flame suppressants. Potassium nitrite proved best (the others: liquid bromine, potassium nitrate, sodium nitrite), made a major reduction in the heat of jet flames. Why this project was apparently dropped was not made clear.

Time Limit: Climaxing first-day activities was the ARS Banquet held Monday evening. Toastmaster Walter T. Olson, president, Cleveland-Akron Section, introduced featured speaker Eger V. Murphree. In his first major address since taking office last April as Special Assistant to the Secretary of Defense for Guided Missiles, Mr. Murphree offered ARS members an optimistic picture of this country's guided missile program. The only limitation on (missile) progress, he declared, is the time it takes to solve technical problems.

Solid Propellant Rockets: J. F. R. Floyd, Applied Physics Lab., Johns Hopkins University, spoke first on "Launching Solid Propellant Rockets from Shipboard Installations" (311-



SIGNING IN, newcomers hurried through registration in order to be on time for the first morning session (ramjets) at 9:30 a.m.



THE AUDIENCES were unusually attentive at most sessions, made many notes for discussion periods that followed talk

ARS MEETING (continued)

56). Mr. Floyd discussed many of the problems peculiar to shipboard use of rockets; e.g., launching from a rolling platform. All such problems have been solved, he concluded; and the future will see growing naval use of solid propellant rockets.

- Joseph W. Wiggins of Thiokol Chemical Corp. talked about the "Importance of Mass Ratio and Adaptability of Case Bonded Solid Propellant Rocket Systems for Achievement of Super Velocities" (303-56). His point, that reduction of inert component weight (as contrasted to improvement of specific impulse) will result in significant improvement of rocket performance, as the author admitted, is still controversial.

- In his paper, "Cold-Forming Methods for Fabrication of Inert Rocket Components During Development" (304-56), H. R. Grant of Aerojet-General described a low-cost, cold-forming process suitable for developmental operations in both aircraft and missile work. In addition to being an economical way of obtaining quantities of parts that would be difficult (or extremely expensive) to make by other means, said Mr. Grant, the process also provides close tolerances, high degree of smoothness in the surface finish, increase in the yield strength and nominal ultimate strength.

- Paper 305-56, "Range, Burnout Velocity, and Design of Solid Propellant Rocket Ballistic Vehicles" by P. J. Blatz and R. D. Geckler of Aerojet-General, was withdrawn because of clearance problems.

Satellite Flight: Lead speaker Ernst Stuhlinger, Army Ballistic Missile Agency, discussed "Instrumentation Problems of Unmanned Satellites" (306-56). In his presentation, Dr. Stuhlinger also investigated the capabilities of satellites for scientific

measurements, offered a tentative scheme for scheduling of planning for satellite instrumentation. A stringent limitation on satellite instrumentation, he pointed out, is the capability of the carrier vehicle. And presently existing missiles, if combined to multi-stage rockets, have satellite capabilities below 100 lb.

- S. F. Singer, University of Maryland, treated a somewhat neglected area of satellite flight in his paper, "The Effect of Meteoric Particles on a Satellite" (307-56). Dr. Singer was concerned primarily with the erosion phenomena due to impact of high velocity meteoric dust. At present, he said, scientists honestly don't know whether or not this dust is really a hazard to satellite flight. But they will have the answer in two years, perhaps.

- A new concept in space vehicles, "The Solar-Powered Space Ship" (310-56), was described by Krafft A. Ehricke of Convair. This ship, as pictured by Mr. Ehricke, would use a nonchemical, solar-powered drive and would have to be lighter ($1/100$ to $1/1000$ lb sq ft) than anything yet designed. Such a vehicle, he admits, goes well beyond the present concept of man-made satellites and is aimed

more for the post-satellite, space travel era. This type ship, he said, is designed to relieve the logistics problem of space travel.

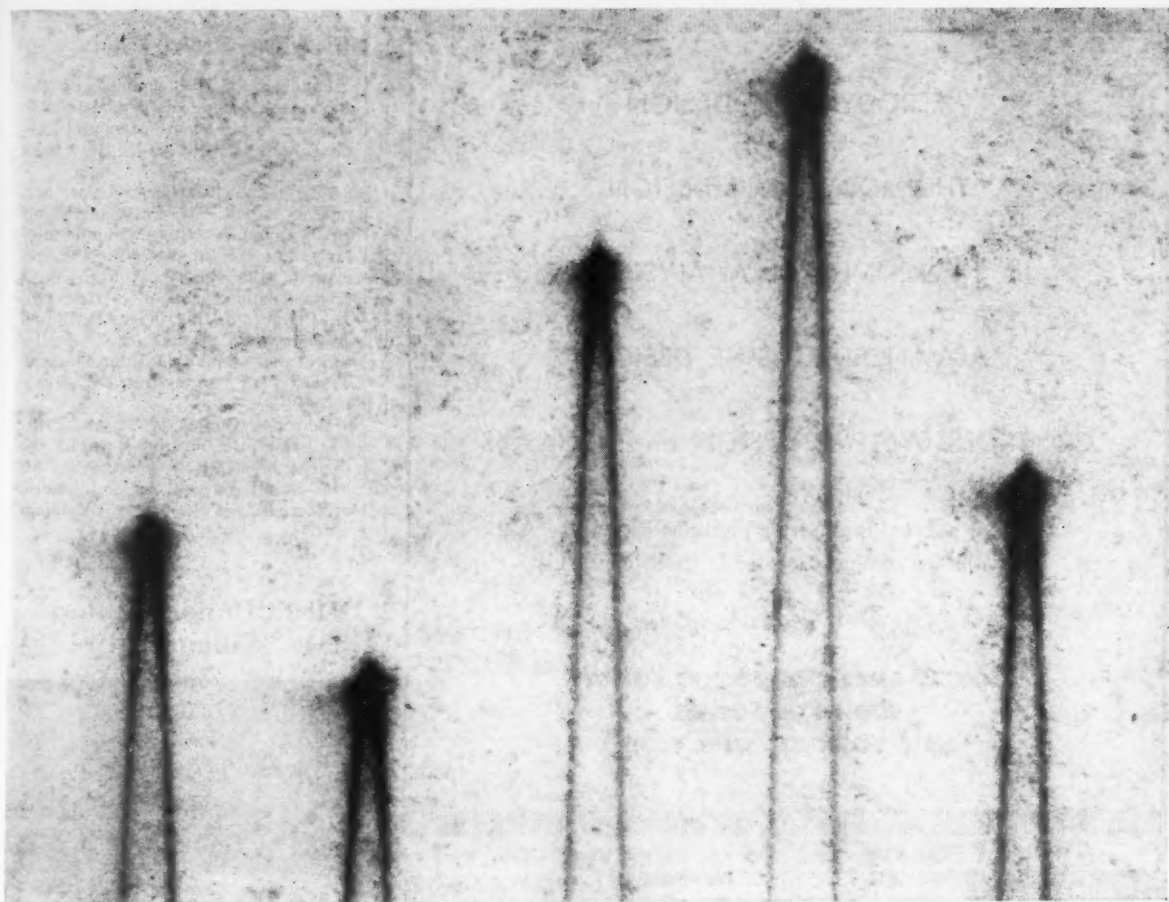
- "Satellite Communication Problems" was the subject of the last paper (309-56), by Jorgen Jensen, The Glenn L. Martin Co. The first question that comes to mind, said the author, is probably the limiting range of a satellite communication system. But range is no particular problem, he concluded; the most serious problems will be equipment reliability and generation of auxiliary power.

- "Lifetime of Artificial Satellites of the Earth" (308-56) by Irvine G. Henry of Aerojet-General was not completed in time for presentation.

Wind-up: Wednesday afternoon, ARS members rode out to the Lewis Flight Propulsion Laboratory of the NACA. Here, after a brief welcome by Abe Silverstein, associate director of Lewis Laboratory, visitors split into four groups, made a three-hour tour of the facility. High spot of the tour was a 20-minute look at NACA's recently opened, "10-by-10" supersonic (up to Mach 3.5), wind tunnel. This visit officially ended the Cleveland Meeting.



INFORMAL GET-TOGETHERS in lobby were common during meeting breaks, gave listeners a chance to go over some of the subjects presented by speakers at sessions



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Fletcher Pratt Dies

Fletcher Pratt, a founder of the American Rocket Society, died June 10 of cancer in Monmouth Memorial Hospital, Long Branch, N. J. He was 59 years old.

A well-known writer, scholar, and military expert, Mr. Pratt gained his greatest prominence as a writer on military and naval subjects. For the first issue of the *Bulletin*, the Society's original publication and forerunner of *JET PROPULSION*, Mr. Pratt contributed a summary of a paper on "The Historical Background of Interplanetary Travel" (*JET PROPULSION*, Nov. 1955, p. 588).

Born near Buffalo, N. Y., Mr. Pratt was living in Highlands, N. J., at the time of his death. Surviving are his wife, Mrs. Inga Stephens Pratt, and a brother, Robert H. Pratt of Washington.

Detroit Honors Keller, Elects Williams



On May 28, ARS National Awards Chairman John Cowen (left) presented K. T. Keller, former board chairman of Chrysler Corp., with a citation for his distinguished service as guided missiles program director for Dept. of Defense from 1950 to 1953. Other meeting highlights were a talk by Lovell Lawrence, Jr. (below, left), Chrysler missile scientist, on the potential use of man-made stars with built-in nuclear power plants as navigational aids, and the election of Charles W. Williams (below right) of Chrysler Missile Operations as Section president.



Section Doings

Cleveland-Akron: At a regular business meeting on May 17, the Section installed the following officers and directors: Adelbert O. Tischler, president; Luiz R. Lazo,



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vice-president; Howard W. Douglass, secretary-treasurer; John C. Feldscher, two-year director. Henry Burlage, Jr., continues as a director for the coming year.

Columbus. On April 26, Section members gathered at Battelle Institute, listened to William Beach, chief thermodynamicist, North American Aviation, Columbus Div., discuss several problems (and some possible solutions) associated with the thermal barrier.

The next meeting, May 24, featured a talk by Fredrick Hitchcock, specialist in aviation physiology at Ohio State Univ. Dr. Hitchcock spoke about the physiological problems associated with space flight.

Connecticut Valley: At a dinner meeting of the M.I.T. Club of Hartford, Section President Charles King, Jr., spoke about "Rockets and the ARS."

Univ. of Connecticut student branch of ARS held an organizational meeting April 18, made plans to complete the organization at the beginning of the fall term. The meeting, moderated by Section Secretary Sheldon Dolinger, featured talks on rockets and the Society by Frank Gabron and Roy Terwilliger of United Aircraft Corp.

National Capital. Mrs. Robert H. Goddard was guest of honor and first speaker at Section's May 23 dinner meeting. Mrs. Jean C. Bergaust, president of the Section's newly formed Women's Auxiliary Group, presented Mrs. Goddard with a life membership certificate in the group.

Following Mrs. Goddard's talk, ARS Board of Directors Chairman Andrew G.

Haley reported on NATO guided missile meeting held in Munich last April; and the Army Ordnance space flight film, moderated by Wernher von Braun, was shown.

New York: On June 9, about 500 members of ARS and American Helicopter Society journeyed to Denville, N. J. Here, Reaction Motors, Inc., played host, provided luncheon and demonstrations of company products. Top items of interest: a new model of the rocket motor that drove the Bell X-1A to world speed and altitude records and a Marine helicopter equipped with ROR (see JET PROPULSION NEWS, p. 583).

Northern California: Leading West Coast scientists (photo) listened to Convair's Krafft A. Ehricke discuss "The Challenge of Space Flight" at a meeting held May 10 at the Engineers' Club in San Francisco. The audience also witnessed the launching of an Aerobee rocket in a film titled "Sky Is No Limit." The program—sponsored jointly by local sections of ARS, ASME, and Institute of the Aeronautical Sciences—capped the Northern California Section's recent membership drive.

Twin Cities: Pending a regular election in the fall, Section officers were appointed on an acting basis at a reorganizational meeting May 11. The appointees: president, C. C. Chang, Univ. of Minnesota Aeronautical Engineering Dept.; secretary, Thomas F. Irvine, Jr., Univ. of Minnesota Dept. of Mechanical Engineering; treasurer, Melvin Diels, Minneapolis-Honeywell Regulator Co.



Oakland Tribune

Watching a Wind Tunnel

Prior to attending the May 10 meeting of the Northern California Section, several scientists visited the new Low Density Wind Tunnel at the Univ. of California's Richmond Field Station. Taking the tour were (above, from left, standing): A. J. Eggers, Jr., president, Northern Cal. Section, ARS; George

Cooper, chairman, Northern Cal. Section, Inst. of Aeronautical Sciences; Krafft A. Ehricke, Convair; F. W. Beichley, chairman, San Francisco Section, The American Society of Mechanical Engineers; (seated) Morrough P. O'Brien, dean, University of California, College of Engineering.

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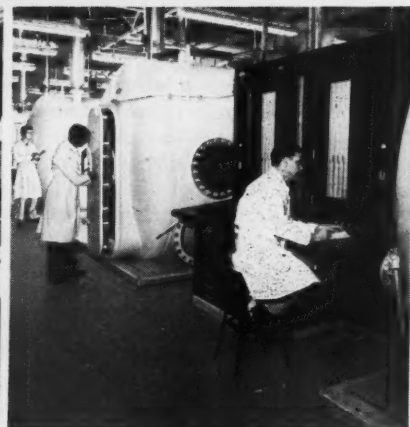
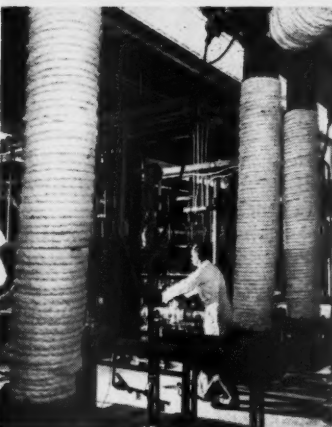
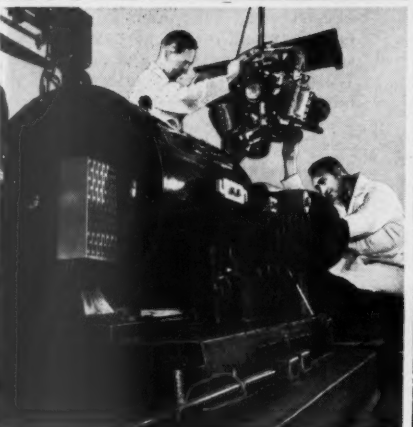
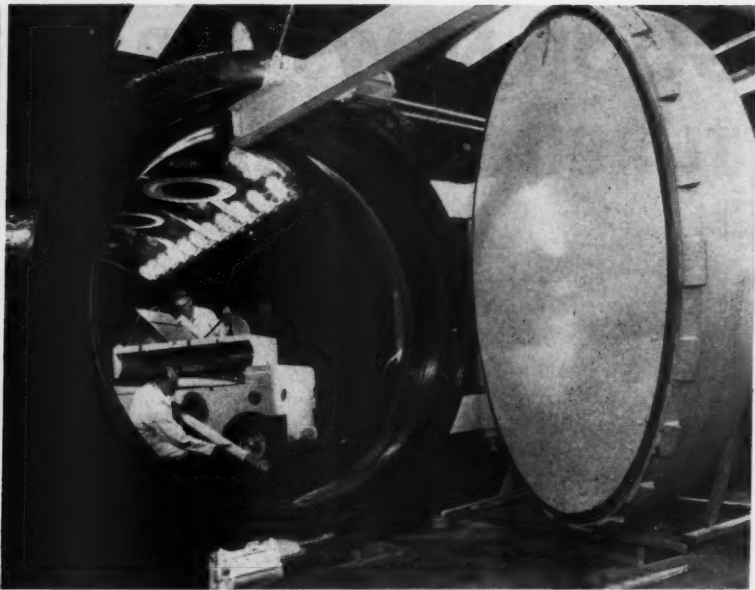
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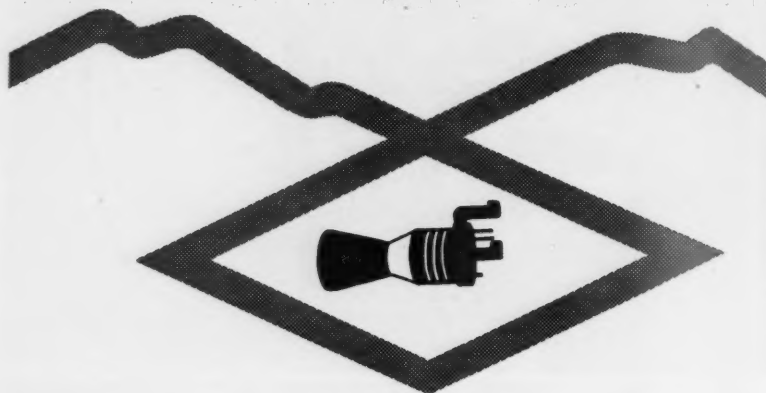
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Attending symposium on history of German guided missiles in Munich April 23-27 are ARS members Arthur P. Adamson (at top, left foreground), James J. Harford (second row, left), and J. B. Cowen (above, second row, right). Meeting was sponsored by AGARD and organized by Wissenschaftliche Gesellschaft für Luftfahrt, the German aeronautical society. Andrew G. Haley, ARS board chairman, was host at a reception for AGARD's chairman, Theodore von Kármán, during the meeting.

. . . and in Montreal



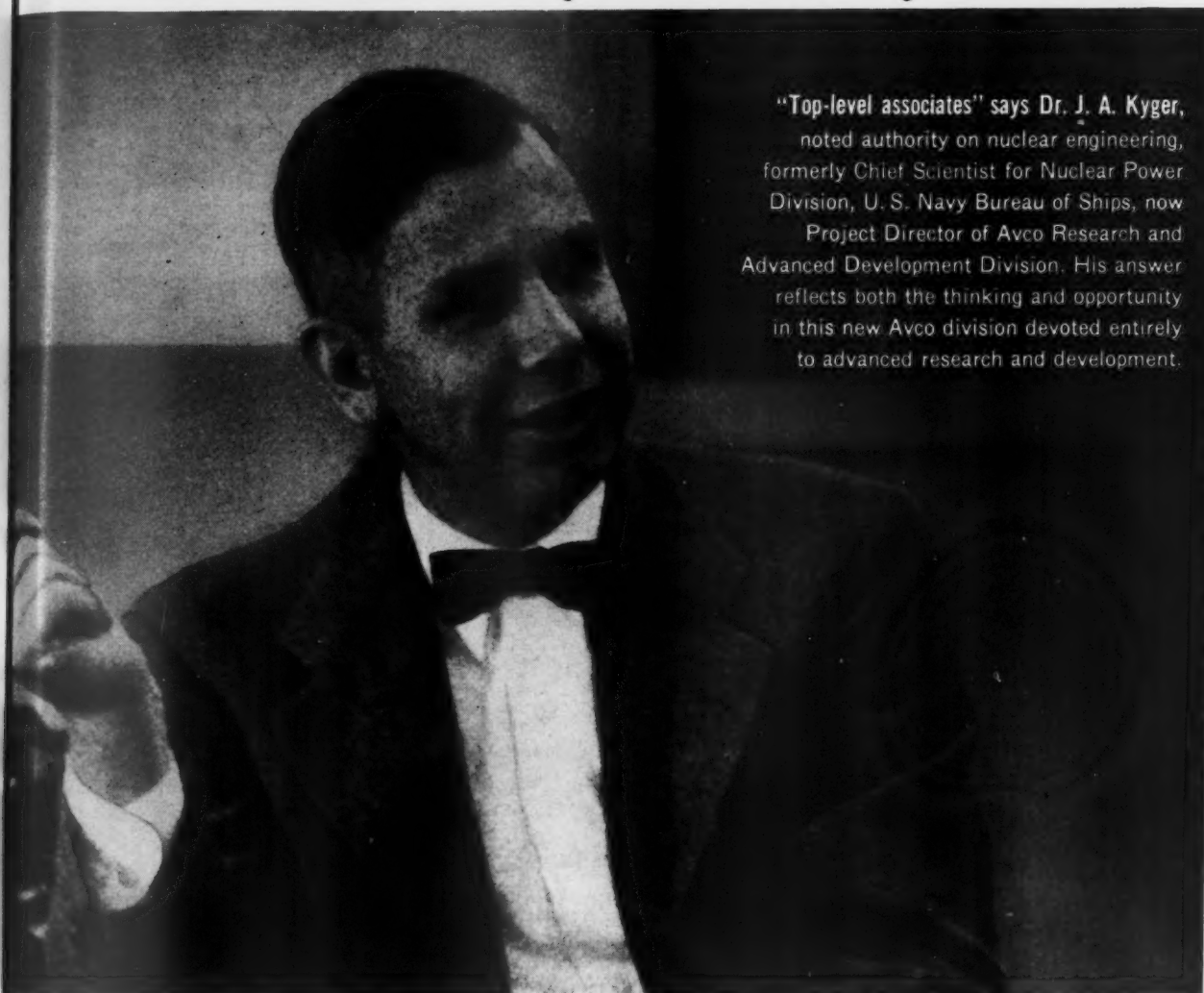
Shown with the president of Canadair Ltd., J. Geoffrey Notman (center) and the president of the Engineering Institute of Canada, R. E. Hertz (right) is ARS Executive Secretary Harford. Occasion is visit to Canadair plant during 70th meeting of EIC which was cosponsored by ARS and ASME. More than 200 engineers attended session in which Admiral F. R. Furth's paper on Project Vanguard, sponsored by ARS, was heard. Meeting took place in Montreal May 23-25.

ARS Meetings Calendar

- Sept. 17-21: Seventh International Astronautical Congress, Palazzo dei Congressi, Rome. Space flight.
- Sept. 17-21: Annual Meeting and Exhibition, Instrument Society of America, Coliseum, New York.
- Sept. 24-26: ARS Fall Meeting, Hotel Statler, Buffalo. Combustion, instrumentation, education, liquid propellants.
- Nov. 25-30: ARS-ASME Annual Meeting, New York. General subjects. (Manuscripts due Aug. 27; to Andrew G. Haley, 1735 DeSales St., Washington 6, D. C.)

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"Solar Soirée"

TOP event of the year's social season in the land of the boondocks was the First Annual Solar Soirée of the New Mexico-West Texas Section. Upon entering the BOQ at White Sands Proving Ground, where the event was held, "space travelers" exchanged tickets for planet passes, checked baggage, weighed in, and awaited flights in the cocktail lounge. Red Fuming Nitric Acid, diluted somewhat with alcohol, was reported to be the principal beverage. Flight arrival and departures were announced, and dancing took place in a star-decorated room.



Left: Space ranger and rocket ship hostess, Mr. and Mrs. James Sims. *Right:* Mr. and Mrs. L. Gardenhire wearing prize-winning costumes



Left to right: Pogo-Hi: Gilbert (Pogo) Moore; Mars-Keteers: Dr. and Mrs. R. K. Sherburne; Parachute: Mrs. Gilbert Moore



Left to right: George W. Gardiner, Capt. Levering Smith, Mr. and Mrs. Herb Karsch, and Nathan Wagner

JET PROPULSION

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Terrier-Armed Cruiser to Defend Navy's Atlantic Fleet Task Forces

BY DAVID A. ANDERTON

Aviation Week, November 7, 1955.—Operational integration of the Navy's first guided missile cruiser—the USS *Boston* (CAG-1)—will begin early next year when the converted ship joins the Atlantic Fleet.

The *Boston* is armed with two twin launchers firing the supersonic Convair Terrier anti-aircraft missile. Fleet operations with the Terrier follow extensive test firings of the missile, including a series from the Atlantic Fleet's experimental gunnery ship, the USS *Mississippi*.

TERRIER HISTORY

The Terrier is the first weapons system to come from the Bureau of Ordnance's ten-year-old Section "T" contract with the Applied Physics Laboratory of Johns Hopkins University. Prompted by Japanese Kamikaze attacks in the closing months of World War II, the bureau established the missile research and development project under the code name of Bumblebee.

Scientists of JHU/APL made initial studies that led to the Terrier; other phases of the Bumblebee program have produced Talos, another anti-aircraft missile with ramjet propulsion, and the Triton.

As Terrier tests made the missile look more and more promising, the Navy opened a new production plant at Pomona, Calif., and turned the operation of the facility over to the Convair Division of General Dynamics. Production began in January 1953.

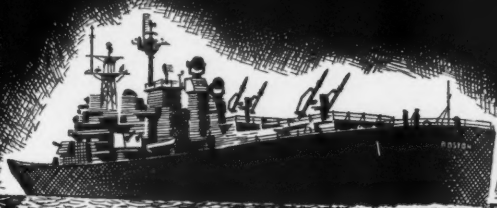
MISSILE DIVISION

The powerplant, a sustainer rocket built by the M. W. Kellogg Co., occupies the bay between the wings and the tail. The aft guidance section mounts the fins, antennas and related avionic gear.

Convair produces both guidance sections, the spines connecting those sections, and all the aerodynamic surfaces.

The *Boston*, as the first guided-missile cruiser, forms the nucleus of the Navy's first guided-missile division.

In addition to actually building the Terrier's powerplant, as mentioned in this *Aviation Week* article, The M. W. Kellogg Company also participated in the initial conception of the Terrier's propulsion system and in the ultimate development of this guided missile as one of the Navy's primary anti-aircraft weapons. To date, M. W. Kellogg's unique industrial background in the utilization and control of high temperatures and pressures has contributed to the engineering and production of over ten different rocket engines for the Armed Forces.



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Manufacturer's Twenty Member Motion Picture Unit sets up to shoot Northrop F-89D all-weather jet interceptor for sequences in Northrop Training Department film.

INDUSTRY'S USE OF 16MM CAMERAS BROADENS

Northrop Aircraft Demonstrates Expanded Industrial Use of Mitchell Cameras

Over 100,000 feet of film were shot last year by two 16mm Mitchell cameras operated by a full-scale motion picture unit at Northrop Aircraft. Operating daily throughout the year, these 16mm cameras provide impressive evidence of the rising role of professional motion picture equipment in American Industry today.

Northrop, a leader in airframe and missile manufacture, makes diversified use of their Mitchell cameras. Motion pictures range from employee activities to engineering test films—where re-shooting is impossible and where steady, accurately-framed film of superior quality is consistently delivered by Mitchell cameras.

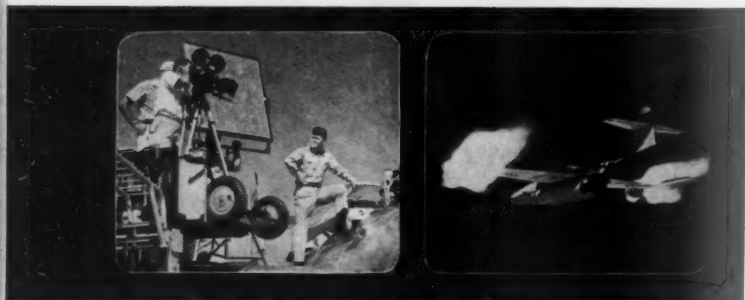
No other single camera is today used by American Industry for such a broad range of filming requirements as is the Mitchell camera. Easy operating Mitchell cameras help create sales, meet delivery schedules, and systematize and accelerate research and development. For details about Mitchell equipment that will meet your specific needs, write today on your letterhead.

For Quality Control Film, Mitchell camera moves in for close shots of Scorpion F-89D.

104 Rocket Salvo of twin-jet F-89D is captured on 16mm Engineering Test film.



Alaska Bound test pilot Bob Love and Columnist Marvin Miles being filmed by Mitchell camera for Northrop Public Relations Department.



Mitchell Camera

CORPORATION

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GLENDALE 4, CALIFORNIA

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*85% of professional motion pictures shown in theatres throughout the world are filmed with a Mitchell

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JULY

New Patents

George F. McLaughlin, Contributor

Blast fence for jet engines (2,726,830) Edward L. Brown, John M. Robertson and George E. Shafer, Middletown, Ohio assignors to Armco Steel Corp.

Spaced parallel curved vanes extending between a metal frame, and set at an angle to the horizontal. The vanes slope in the opposite direction to the framework to deflect the blast.

Pyrotechnic compositions. Raymond H. Heiskell, San Bruno, Calif.

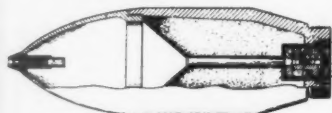
Dark burning no-flash ignition pellets for igniting a delayed ignition projectile tracer with up to 10% of lubricant.

Pellet (Patent 2,726,943) comprises at least 10% not to exceed 90% of an oxygen bearing compound selected from the group consisting of cupric oxide, barium peroxide, antimony pentoxide, lead dioxide and lead chromate.

Pellet (Patent 2,726,944) comprises 90-30% barium peroxide 10-70% of a sulfide selected from the group consisting of chromium monosulfide, chromium tetrathiolate, and selenium sulfide.

Fuel for and method of operating a jet engine (2,729,936). Sylvester C. Britton, Bartlesville, Okla., assignor to Phillips Petroleum Co.

A fixed size fuel consisting of between 5% and 95% by volume 1-olefins boiling between 90 and 500 F is injected into the combustion chamber. Air is injected into the chamber at a Mach number between 0.01 and 1.00 at a pressure between 0.2 and 40 atm, at a temp between -30 and 1040 F. Fuel is burned at a combustion efficiency within range from 40% to 100% to heat the air and gases.



2,730,046

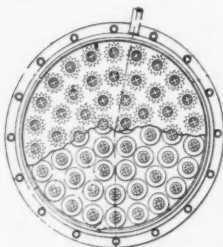
Safety device for the explosive head of a missile (2,730,046). Lars O. Bergstrom and Karl E. W. Hjelm, Bofors, Sweden, assignors to Aktiebolaget Bofors Corp.

Fuze magazine for detonating the charge of a rocket-type missile. The force of the initial linear acceleration by the missile in flight acts upon a yieldable means within the explosive head responsive to a physical magnitude developed and rendered effective upon discharge of the missile.

Method of and apparatus for guiding flight bodies along preselected flight paths (2,730,715). Gustave Guanella, Paul Guttinger, Max Lattman, and Georg Weber, Zurich, Switzerland, assignors to "Patelhold" Patentverwertungs and Electro-Holding A. G. and "Contraves" A. G.

Flight bodies are steered along a desired flight path in accordance with the degree of modulation of signal energy received on the flight body. Two individually distinctive beams of ultrashort-wave radio energy and of different angular spread, are employed. The beam of greater spread is rotated about the desired flight path with its axis of maximum intensity at a preselected relatively large angle, and the beam of lesser spread about the desired

flight path is rotated with its axis of maximum intensity at a different and relatively small angle.



2,733,570

Injection head for jet propulsion system (2,733,570). George L. Macpherson, Scotia, N. Y., assignor to General Electric Co.

Sheet metal outlets in an injection head, having hollow conelike protusions, each terminating in a face. A flat sheet-metal faceplate has holes mating with the faces surrounding the faces for the issue of fuel reactants.

Jet-driven helicopter rotor power plant control system (2,734,585). Richard H. Ball, Thomas B. Murdock, and Hugh M. Ogle, Schenectady, N. Y., assignors to General Electric Co.

Control system for a rotor-tip jet propelled helicopter. Control lever with a fulcrum variably positioned to control fuel-air ratio supplied to rotor tip jets in response to movement of main regulator.

Airplane control surface and jet deflector arrangement (2,734,698). Joseph W. Straayer, Birmingham, Mich., assignor to Boeing Airplane Co.

Jet engine located beneath a control surface at the trailing edge of an airfoil. The surface is capable of movement from a normal neutral position into a controlling position. The rear end of the tailpipe located ahead of and sufficiently close to the surface so that the undeflected jet from the tailpipe impinges on the control surface or jet will clear surface with movement into its downward deflected position.

Support means (2,735,262). Frederick L. Geary, Springfield, Mass., assignor to United Aircraft Corp.

Afterburner with variable area nozzle surrounding the thrust nozzle and movable into a position for decreasing the effective area. When the thrust nozzle is in operation it expands into engagement with the ring surrounding the thrust nozzle, the space between the nozzles sealed to minimize flow of hot gas past the ring.

Combination rocket and ramjet power plant (2,735,263). Jack O. Charshafian, Englewood, N. J., assignor to Curtiss-Wright Corp.

Rocket carried in an open ductlike member in a jet engine. The rocket has a combination chamber and a conical conduit which discharges rocket combustion gases into a hollow conical jet to induce airflow through the duct.

Nozzle construction for jet engines (2,735,264). Charles S. Jewett, Ramsey, N. J., assignor to Curtiss-Wright Corp.

Adjustable nozzle comprising two pairs of concentric movable spherical segments. Each inner pair of segments has a smaller

Miniature Pressure Transducers

STATHAM UNBONDED STRAIN GAGE TRANSDUCTION

MINIMUM RESPONSE TO VIBRATION OR ACCELERATION

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20 MILLIVOLTS AT 5 VOLT EXCITATION

PRESSURE ADAPTERS FOR CLOSED LINE APPLICATIONS

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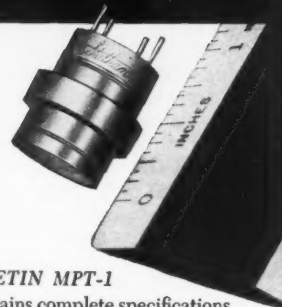
NO EPOXY RESIN PRESSURE SEALS

MAXIMUM LINEARITY

ABSOLUTE PRESSURE 0-5 to 0-150 PSIA... MODEL P130

DIFFERENTIAL PRESSURE ± 2.5 to ± 25 and 0-5 to 0-150 PSID... MODEL P131

GAGE PRESSURE 0-5 to 0-150 PSIG... MODEL P132



BULLETIN MPT-1

contains complete specifications on the foregoing pressure transducer models.

All matters pertaining to sale or use of instruments of our manufacture are handled by engineering personnel directly from our Los Angeles plant.

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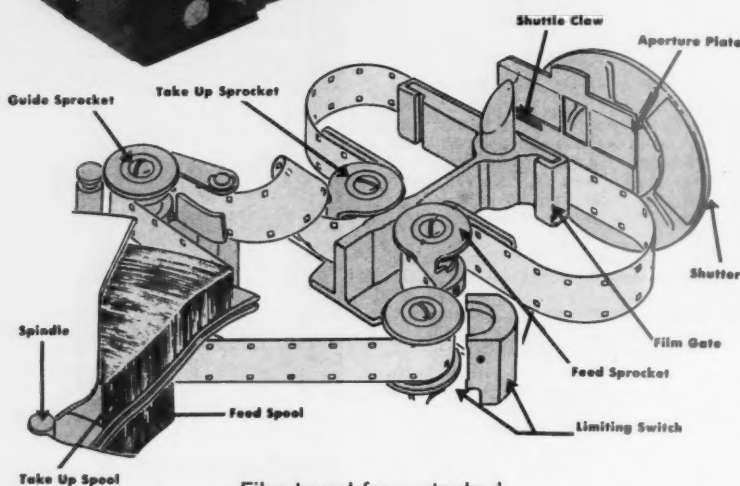
EDITOR'S NOTE: The patents listed above were selected from recent issues of the Official Gazette of the U. S. Patent Office. Printed copies of patents may be obtained at a cost of 25 cents each, from the Commissioner of Patents, Washington 25, D. C.

NEW HIGH-SPEED CAMERA

Developed to record accuracy
of ground-to-air and air-to-
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Traid 500 16mm
Motion Picture Camera



Film travel from stacked
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TECHNICAL DATA AND OPERATING CHARACTERISTICS:

FILM SIZE: 16mm...**FILM CAPACITY:** 200 ft. daylight loading spool...**DIMENSIONS:** 6½" high x 5¼" wide x 7-5/8" deep...**MOTOR:** 28V-DC or 115V-AC (optional)...**POWER CONSUMPTION:** 28V=5 amps—115V=1.4 amps...**CAMERA SPEEDS:** 100 and 200 fps standard...**LENS MOUNT:** Single lens "C" screw type. GSAP Mount available...**EXPOSURE:** 1/1000 sec. at 200 fps with 72° shutter...**CAMERA MOUNT:** Dovetail plate mates with permanent mount receptacle.

OPTIONAL FEATURES:

POSITIVE VIEWFINDER: For direct viewing...**CORRELATION SWITCH:** Provides output pulse for every 40 frames...**TIMING SYSTEM:** Places 120 or 60 pips per sec. on film...**EVENT TIMER:** To control timing system to record an event...**LIMITING SWITCH:** Cuts off power at end of run...**HEATER:** Thermostatically cuts in at +60F...**PILOT PIN:** Provides positive frame to frame registration.

ACCESSORIES:

RADIO INTERFERENCE FILTER: In power line...**BORESIGHT:** Two types available...**ERECTOR ASSEMBLY** for 90° viewing...**TRACKING FINDER:** 16mm reticle with calibrations for lenses from 0.7 to 4.0 in...**INSULATION BLANKET:** For excessive heat or cold...**RECEPTACLE:** Mating permanent dovetail plate...**HUBER MOUNT:** CARRYING CASE: LENSES.

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radius than the outer pair so as to be pivotally movable for varying the exhaust duct area.

Warship weapons system, including aircraft storing and launching arrangement (2,735,391). Herbert H. Buschers, Towson, Md., assignor to The Glenn L. Martin Co.

Cylindrical hangar for storing rockets or aircraft on the deck of a submarine. Rockets, stacked one above the other on guide rails, may be moved to a launcher which is locked in the raised position for launching, and retracted against the deck when not in use.

Jet propelled aircraft (2,735,633). James W. Manning, Huntington Park, Calif.

Fuselage having a pointed nose with bottom and top surfaces rounded and side portions flat. A pair of spaced passageways for air intake extend through the body and communicate with spaced cut-outs in the tailpipe. Sprayed fuel is burned with air from the passageways to produce additional thrust.

Afterburner and exhaust nozzle controls for gas turbine engine (2,736,166). Frank C. Mock, St. Joseph, Ind., assignor to Bendix Aviation Corp.

Tailpipe terminating in a reaction jet. The jet area may be varied to maintain constant preselected pressure ratio between compressor inlet pressure and tailpipe pressure during all conditions of engine operation.

Air fuel ratio control for ramjets (2,736,167). Walter D. Teague, Jr., and Pasquale A. DePadova, Newark, N. J., assignors to Bendix Aviation Corp.

Temperature sensing means at the inlet and outlet conduits of the combustion chamber control the quantity of fuel pumped into the chamber. A fluid pressure turbine drives the variable speed fuel pump.

Liquid fuel control system for a rocket engine (2,738,648). Hugh M. Ogle, Schenectady, N. Y., assignor to General Electric Co.

Rocket propelled fuel system with a bellows responsive to pressure output of a liquid oxygen pump. A control regulates the flow of oxygen to the combustion chamber, and manual means are provided for adjusting the effect of the bellows.

Gas turbine engine with thrust balancing coupling (2,738,920). Chas. J. McDowall, New Augusta, Ind., assignor to General Motors Corp.

Means for opposing end thrust to reduce thrust bearing load. An elastic member between the stator and rotor is coupled to be stressed by thermal expansion, exerting a thrust on the rotor in opposition to the rotor end thrust.

Boundary layer control apparatus for compressor (2,738,921). George F. Hausmann, Hartford, Conn., assignor to United Aircraft Corp.

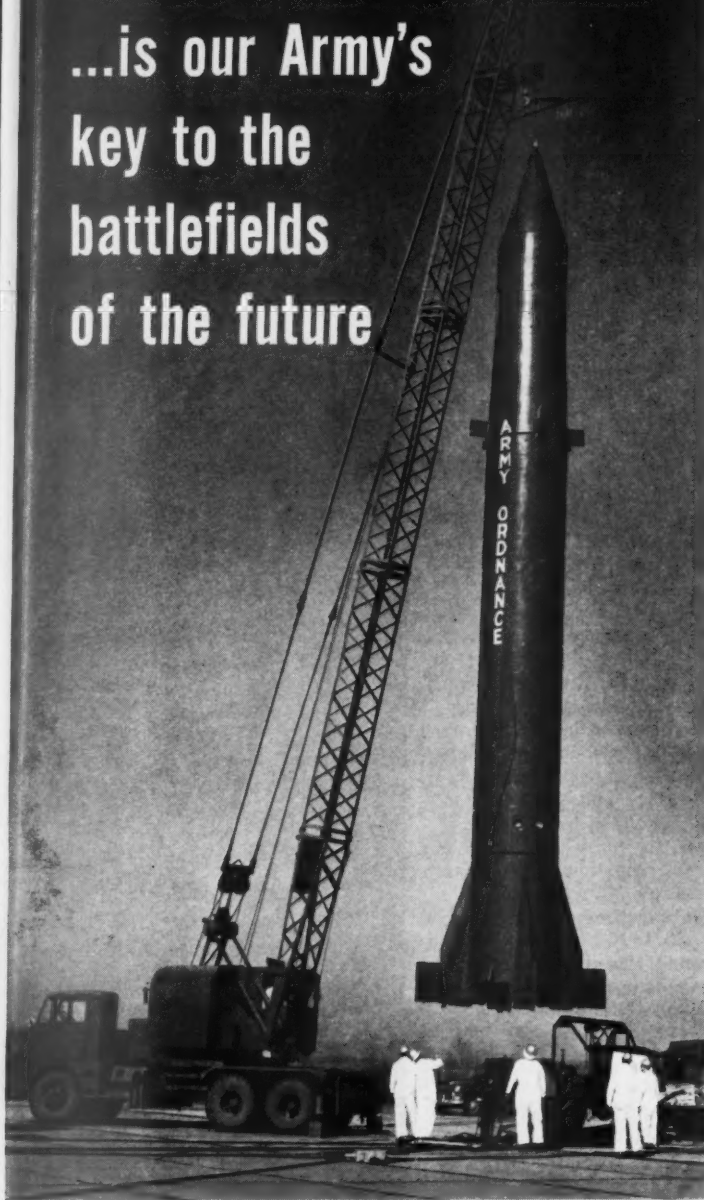
Scoop adjacent to the inner radial end of stator blades for inducting boundary layer flow along the inner wall of an annular passage. The flow is discharged into the passage adjacent to the outer wall of the passage.

Temperature responsive control system for gas turbine power plant having exhaust reheating (2,739,441). John H. Baker and John W. Jacobson, Schenectady, N. Y., assignors to General Electric Co.

Tail pipe reheat fuel burning for thrust augmentation. Variation from a predetermined desired temperature is corrected automatically by trimming the reheat fuel regulator.

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of the future



U. S. Army Photo

This is one of a series of ads on the technical activities of the Department of Defense.

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JULY 1956

● One of eight permanent Ordnance Corps arsenals, Redstone is the center for the Army's rocket and guided missile program. Its 40,000 acres are located on the Tennessee River near Huntsville, Ala., and house research laboratories, environmental test equipment and several rocket-testing ranges. Redstone Arsenal's military and civilian scientists and engineers produce weapons ranging from a tiny training rocket to the giant IRBM, now being developed on a crash basis by the recently-established Army Ballistic Missile Agency.

From the research, development, production and field service headquarters located at Redstone flow thousands of directives covering the rocket and guided missile work being done by research laboratories, universities and private industry throughout the nation. Weapons systems developed by this Ordnance-Industry team include the Super Bazooka infantry rocket, the Honest John artillery rocket, the Corporal missile and the Nike anti-aircraft missile.

Scientific barriers of all kinds are being broken by the 9,000 employees of Redstone Arsenal, but the exciting and difficult technology of guided missile development constantly presents new problems. Electronic computers click away at missile trajectory formulas by day, while at night rockets equipped with headlights streak down-range, adding valuable data to our country's newest arsenal of defense.

REDSTONE BALLISTIC MISSILE — This long-range rocket-powered, bombardment weapon was developed by the arsenal's guided missile team headed by Dr. Werner von Braun. The Redstone is the progenitor of the Army's IRBM, the Jupiter.

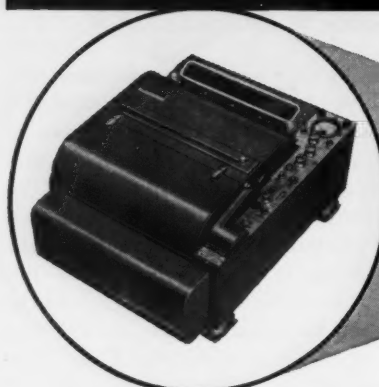


FORD ENGINEER checking voltage and frequency accuracy of power supply unit under simulated load conditions in a project for the Guided Missile Development Division of the Redstone Arsenal.

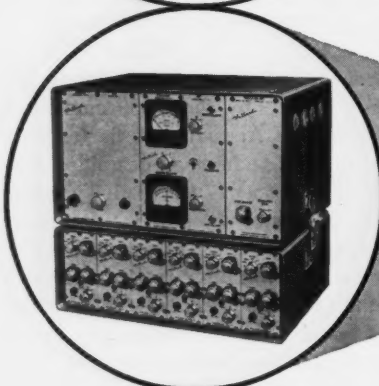
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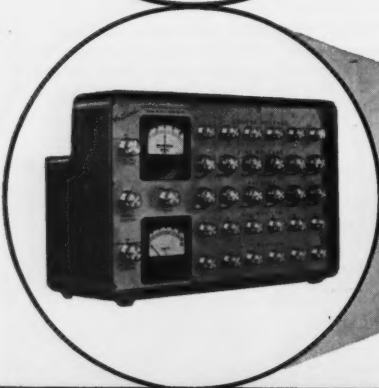
- VERSATILITY • PERFORMANCE
- EASE OF OPERATION
- FOR RELAY RACK OR TABLE MOUNT



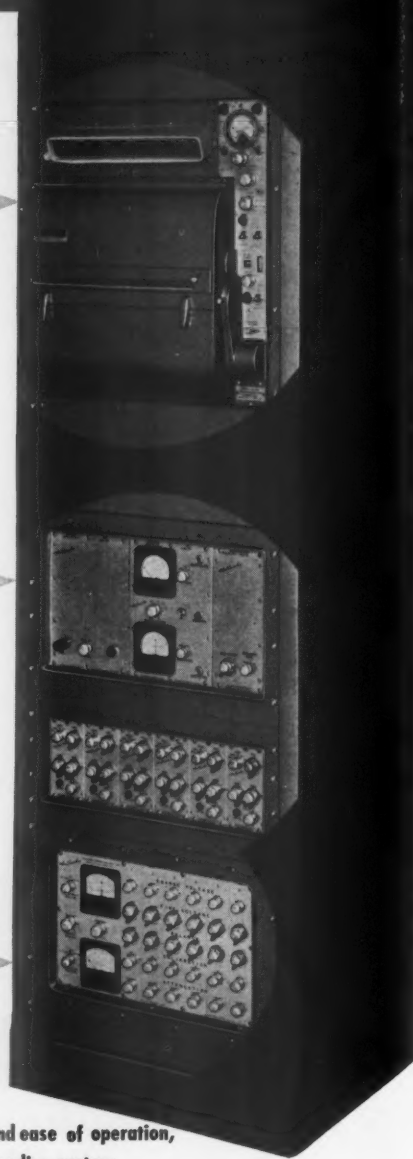
The Series 700 oscillographs feature 8" paper width with 1-36 channels, or 12" paper width with 1-60 channels. Available for 28 v.d.c. or 115 v.a.c. operation, the 700 Series has paper speeds adjustable from .030 to 144"/sec., and writing speeds in excess of 20,000"/sec. Separate supply and take-up drums are light-weight—and light tight for easy daylight loading.



The Heiland 119 Amplifier System offers up to 6 channels, in any combination, of either linear-integrate amplifiers or carrier amplifiers. Carrier amplifier channels provide linear frequency response from 0 to 1000 CPS, for resistive, linear differential transformer, or variable reluctance type transducer inputs. Linear-integrate amplifier channels provide linear frequency response from 5 to 3000 CPS for self-generating transducers. Provides high-amplitude recording up to 8" peak to peak deflection.



The Heiland 82-6 Bridge Balance and Strain Indicator Unit provides a simple and accurate means of balancing, calibrating and measuring static and dynamic phenomena from resistive-type transducers where you don't need amplification. When used as a strain-indicating device without an oscillograph, an input of 25 microamperes produces full scale on the indicating meter.



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New Equipment and Processes

Equipment

Electrical, Electronic

Resistor Network. Type 1299 is plug-in, epoxy molded, 9-pin miniature socket. Daven Co., 530 W. Mt. Pleasant, Livingston, N. J.

Printed Circuit Connector. Series 4PCSC available for 22 contacts. Electronic Sales Div., DeJur Amco Corp., 45-01 Northern Blvd., Long Island City 1, N. Y.

Power Resistor. Type PW-5 miniature 5 w, high temp, rectangular design. Fuse Resistor. Type FS functions as resistor under normal load conditions and fuse under abnormal conditions. International Resistance Co., 401 N. Broad St., Philadelphia 8, Pa.

Counterpoise Relay. Tandem type, 24-pole double throw, 2 amp, 28 v dc with total weight of 10½ oz. Cook Electric Co., 2700 Southport Ave., Chicago, Ill.

Aluminum Dry Battery. Replacement for zinc cell has long shelf life. Potential of 1.6 v or 1.7 v. Aluminum Corp. of America, 1501 Alcoa Bldg., Pittsburgh 19, Pa.

Test

Fatigue Tester. Multirange model has capacity of 120,000 lb. Ivy Co., Norwalk, Conn.

Compression and Tension. Tester has capacity of 60,000 lb and features stress-strain recorder. Baldwin-Lima-Hamilton Corp., Philadelphia, Pa.

Transistor Tester. Model T-62 is direct reading. 50 w capacity. 5 current ranges from 50 to 500 ma. Scientific Specialties Corp., Snow & Union St., Boston 35, Mass.

Electrostatic Charge Meter. Aimed like pistol, determines charge and polarity. Custom Scientific Instruments, Inc., Kearny, N. J.

High-Speed Camera. Triad Model 500 features 200 fps single speed. Full speed in 1/3 sec., 200 ft capacity. 28 v dc or 115 v ac. Triad Corp., 4515 Sepulveda, Sherman Oaks, Calif.

Surface Thermometer. For spot checks to 1000 F. Pacific Transducer Corp., 11836 W. Pico, Los Angeles 64, Calif.

Running Torque and Dynamometer. Type 200 for testing motors of 1/10 to 1/4 hp. Torque capacities of 1 in.-lb to 5 in.-lb with accuracy of 1/2%. John Chatillon & Sons, 85 Cliff St., New York 38, N. Y.

Instrument Console. For environmental test equipment. Tenney Engineering, 1090 Springfield Rd., Union, N. J.

Creep Tester. 20,000-lb capacity with furnace for tests to 1800 F. Baldwin-Lima-Hamilton Corp., Philadelphia 42, Pa.

Wavelength Tester. Mercury 198 source in portable unit. 12.2 cm wavelength in 2400-2500 megacycle band. Output of 125 w. Accuracies of 1 part in one million. Baird Associates, Inc., 33 University Rd., Cambridge 38, Mass.

Portable X-Ray. 200 kv inspects 3 ft per min exposure. Detects defects or corrosion. Holger, Andreasen, Inc., 703 Market St., San Francisco, Calif.

Ultraviolet Microscope. Model UV-91 for structure determination and indication of absorption spectrum. Resolutions of

0.1-0.2 micron in full color. Boston Electronics Div., Norden-Ketay Corp., Snow & Union St., Boston 35, Mass.

Nondestructive Leak Detector. For hermetically sealed parts and containers. Reed-Curtis Nuclear Industries, Inc., 655 W. Washington Blvd., Los Angeles 15, Calif.

Radiation Measurement. Model HC amplifier and detector for infrared, visible, and ultraviolet spectrum. Baird Associates, 33 University Rd., Cambridge 38, Mass.

Electron Diffractograph. For direct observation or photo recording. 20-50 kv beam. Specimen sizes of 1 in. and sample tests of up to 1000 C are possible. Norden-Ketay Corp., 555 Broadway, New York 12, N. Y.

Oscillograph Camera. For 3-in. CRT. Type 339 offers full binocular vision, is self-supporting and produces prints one minute after exposure. Shutter speed of time, bulb, 1/100, 1/50, and 1/25 sec. Allen B. DuMont Laboratories, Inc., 750 Bloomfield Ave., Clifton, N. J.

Mechanical

Flow Transmitter. Mercuryless, pneumatic balance. Transmits readings up to 1000 ft away. Range of 0-20 or 0-200 in. of water at 0.1% full scale sensitivity. Pressure ratings of 750-1500 psi.

Mechanical Flowmeter. For differential pressures of 53, 119.25 or 212 in. of water. Pneumatic Pressure Controller. Ranges of 0-200 psig, accuracy of ±1%. Tel-O-Set Recorder. Indicator and controller for temperature, pressure, flow, and liquid level. Minneapolis-Honeywell Regulator Co., Wayne & Windrim St., Philadelphia 42, Pa.

Selector Valves. Solenoid-operated 3-way for 1/4-1/2 in. tube sizes, 3-6 gpm flow rates. **Solenoid-Operated 3-Way Valve.** Series 6000 has operating pressure of 3000 psi, 18-30 v dc, -65 to +160 F. Aircraft Products Co., 300 Church Rd., Bridgeport, Pa.

Two-Way Valve. Straight-through, leakproof, for acids, etc. Stockdale Engineering Co., Haddonfield, N. J.

Straddle Carrier. 30,000-lb capacity, 131 hp gives speed of 38 mph. Clark Equipment Co., Ross Carrier Div., Benton Harbor, Mich.

Pressure Gage. 1-in. size for missile use. Ranges to 5000 psi, accuracy of ±2.5%, weight is 0.15 lb. United States Gage Co., 000 Clymer Ave., Sellersville, Pa.

Processes

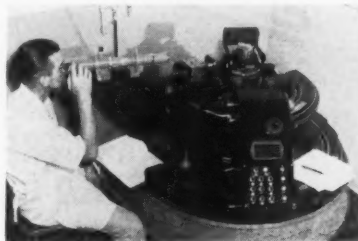
Working with Titanium. At a recent Solar Aircraft symposium on titanium it was emphasized that some alloys have been in mill production for less than one year. Some difficulties are: bend strength; formability of sheet material is not yet uniform. Suppliers are working on rolling techniques so that variance will be less than 5-8% of true flatness. Analytical standards for titanium and its alloys are yet to be established. Special shapes and types are not yet rapidly obtained. Suppliers are still on the "learning curve." Prices of finished materials range upward from \$20 per lb. In engine design, however, it was pointed out that 1 lb saved in a jet engine allows a savings of 10-15 lb in structure.

Ceramics for Rockets. Silicon-nitride bonded silicon carbide as developed by the Refractories Division of Carborundum Co., Perth Amboy, N. J., is finding uses in rockets beyond nozzles (photo). Dimension tolerances of ±0.005 in. per in. of dimension can now be maintained. Special shapes (even including internal and external threads) are possible and oxidation resistance is very high. Possible applications include pump parts, acid spray nozzles, liners, etc.



Speedy Sealing. Small collapsible tubes are being used effectively to apply sealing compounds in aircraft and missile manufacture. This technique makes application less messy, saves material, and eliminates atmospheric contact.

Snark Accuracy. Tolerances on some machining operations on the Snark missile (Northrop Aircraft Co., Hawthorne, Calif.) have had to be held to as low as 30 sec of arc; in some cases, 1 sec of arc. Usual one degree of tolerance on conventional aircraft could not be used. Reason is to insure a minimum amount of error in guidance and critical components. For example, at a range of 5000 miles, tolerances as set above would allow the missile to come within 120 ft of its intended target. Optical precision is called for (photo). Granite foundations are needed for machining precision gears. Tolerances are 0.1 sec of arc. Milling produces parallelism of 0.0002 in. in a 10-in. span.



Product Literature

Asbestos Safety Clothing. 11-page booklet describes line of asbestos suits, helmets, gloves, aprons, leggings, spats, shoes, blankets and curtains. Johns-Manville, 22 E. 40th St., New York 16, N. Y.

New Shop Techniques. Techniques and devices developed at USAF Cambridge Research Center are described in "New Shop Techniques and Developments," PB 111668. Office of Technical Services, Dept. of Commerce, Washington, D. C. \$0.75.

Noise. Human behavior under high intensity noise is described in "The Effects of Noise on Human Behavior," PB 111402. Office of Technical Services, Dept. of Commerce, Washington, D. C. \$2.

Book Reviews

Ali Bulent Cambel, Northwestern University, Associate Editor

Conduction Heat Transfer, by P. J. Schneider, Addison-Wesley Publication Co., Inc., Cambridge, 1955, 395 pp. \$12.50.

Reviewed by ROBERT BROMBERG
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This book contains a surprisingly complete review of the techniques of solution of conduction heat-transfer problems. It is concerned with solutions of the heat conduction equation when applied to various geometries and including both steady-state and transient problems and distributed heat sources. The chief value of the book lies not in the fact that there equations are solved, but that the difficult details of solution—when geometry is complex and boundary conditions not simple—are presented in an understandable manner.

Most of the important mathematical procedures are restated so that the reader finds a minimum of background necessary. The standard methods of solving linear partial differential equations are reviewed and then applied to the problems of interest. Following this, the mathematics appropriate to the numerical solution of conduction problems is presented so that the basis for such solutions is available to the reader. The order of

subject matter is as follows: Steady one-dimensional systems, extended surfaces and steady two-dimensional systems, with appropriate mathematics interspersed; steady state problems using numerical methods; steady heat source systems and porous systems; transient systems including numerical methods; experimental analogic methods. In all cases an ample number of examples are worked out in detail.

The author states: "The reader who is already conversant with heat-conduction literature will appreciate that this present text has little claim to originality." However, in this reviewer's opinion the author has done an excellent job of bringing together, in readable form, the material relating to conduction. About the only omission worth noting is a lack of emphasis on variable property problems. Situations in which both thermal conductivity and heat capacity vary appreciably with temperature (and including phase changes) are of considerable importance in many industrial applications; such problems could be profitably included in the sections on numerical methods (transient in particular) and on analogs.

It is noted in the Preface that the book is intended as a text for graduate-level courses and as a supplementary text for undergraduate courses. It will prove a

useful addition to the library of the heat-transfer specialist concerned with conduction problems.

Preview of Second AGARD Combustion Book

by ALI BULENT CAMEL

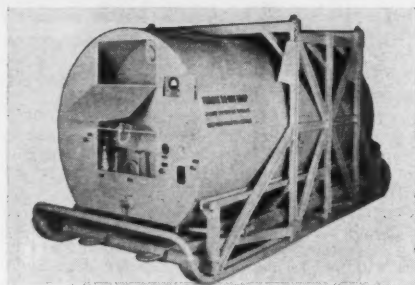
Although it is customary to review published books in these columns we shall deviate once in a while to give our readers prepublication reviews. We shall set this precedent by perusing the papers which were presented at the Second AGARD Combustion Colloquium held in Liege, Belgium, December 5-9, 1955.

The papers which will eventually be brought together as an AGARDograph and their respective authors are: "Scaling of Liquid Fuel Rocket Combustion Chambers," by Chandler C. Ross; "Fuels for Turbojet Powered Aircraft," by R. R. Hibbard and H. C. Barnett; "Fundamental Principles of Flammability and Ignition," by B. Lewis and G. von Elbe; "Problems of Combustion Under Altitude Condition (Treated From Fundamental Viewpoint)," by H. G. Wolfhard; "Quelques Considerations Sur La Convection De La Chaleur Aux Grandes Vitesses et Aux Temperatures Elevees," by E. A. Brun; "Thermal Ignition, With Particular Reference to High Temperatures," by R. S. Brokaw; "Recherche Du Bruleur Optimum en Ecoulement Air-Kerosene a Grande Vitesse," by B. Salmon and H. Vigne; "Inflammation et Allumage Dans Les Moteurs-Fusees a Propergols Liquides," by M. Barrere and A. Moutet; "Les Performances Des Moteurs D'Avions," by A. Capetti; "Combustion en Altitude Dans Les Turboreacteurs," by J. Soissons; "Scaling of Gas Turbine Combustion Systems," by D. G. Stewart; "Sources of Transport Coefficients and Correlations of Thermodynamic and Transport Data," by J. Hilsenrath; "Combustion in the Turbojet Engine," by S. Way; "The Influence of Altitude Operating Conditions on Combustion Chamber Design," by S. L. Bragg and J. B. Holliday; "Similarities in Combustion, a Review," by A. E. Weller; "The Ignition of Flowing Gases," by L. D. Wigg; "Fundamental Equations in Aerothermochemistry," by Theodore von Karman.

It is significant that at least three papers were presented on the rather controversial and complex subject of scaling and similarity. In a review paper, Weller discusses the status of the field. Once again it is brought out that the problems of scaling which for individual components are occasionally tractable become ambiguous when complete systems with multifarious effects are considered. Stewart ably examines scaling in gas turbine combustion systems and shows by limited experimental data (for gutter and cone stabilizers, for example) that correlation between a full-size unit and a one-third size model operated at three times the pressure but the same velocity is quite good. The derivation of significant di-

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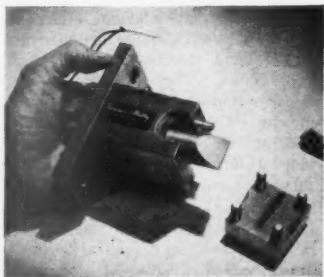
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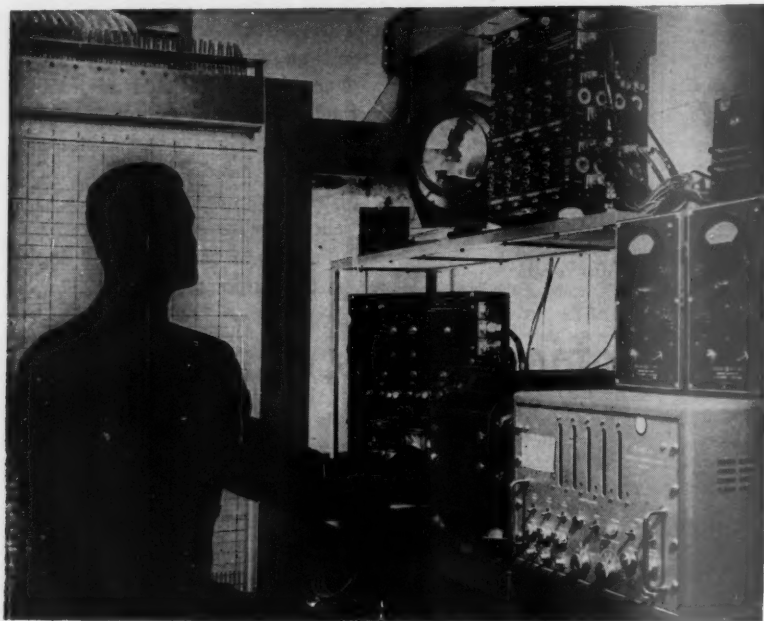
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mensionless parameters as well as the correlation of other experimental data in this chapter make it most significant and helpful to the development engineer. Ross in dealing with scaling in liquid fuel rocket motors suggests that the main reason for the very limited understanding of similarity is due to instability phenomena. In a comprehensive discussion Crocco analyzes existing scaling rules and makes some very significant remarks. These last two contributions are indeed most helpful to the person dealing with instability as well as with scaling in rocket motors. Lewis and von Elbe discuss the principles of flammability and ignition from a phenomenological viewpoint and expressly avoid the use of mathematics in an attempt to better understand the phenomena. They review some interesting observations and reiterate their belief in the excess enthalpy concept. They substantiate their thinking with experimental data but their suggestions require further interpretation in the light of strong conflicts with other observations. In a very elegant analysis, von Kármán develops the fundamental equations of aerothermochemistry. He discusses the meaning of the parameters related to chemical reactions and once again manifests his fine insight into scientific problems. This chapter serves as an excellent guide in relating empirical observations to analytical studies. Thus it is most significant because only when analysis and empiries are properly interrelated will aerothermochemistry be adequately understood.

In a contribution on high velocity combustion Salmon and Vigne use the sigma function as a criterion of stabilization. In two papers Brokaw and Wigg discuss ignition phenomena. Brokaw compares extensive data of ignition delay. Wigg considers the ignition of flowing gases. These two authors ably review the subject and their efforts together with that of others should do much in a more rational understanding of ignition phenomena, something badly needed.

Wolfhard reviews combustion phenomena at high altitudes and discusses the important influence of pressure on combustion. His paper includes empirical observations concerning both premixed and diffusion flames. Another basic problem pertinent to jet propulsion is aerodynamic heating and some of the existing equations are reviewed by Brun for so-called high speeds and elevated temperatures.

A more detailed knowledge of transport data and thermodynamic parameters is absolutely necessary in studying combustion. Hilsenrath reviews the work accomplished by himself, his associates, and others. In a series of chapters, Way, Barrère and Moutet, Soissons, Capetti, Hibbard and Barnett, Bragg, and Holliday discuss combustion phenomena under various circumstances in engines. The information and interpretations in these chapters should serve as an invaluable guide to the engine designer.

It is gratifying that the authors discuss their topics in the broad sense. This approach makes the contributions invaluable to both the experienced worker in the field and to the newly initiated one.

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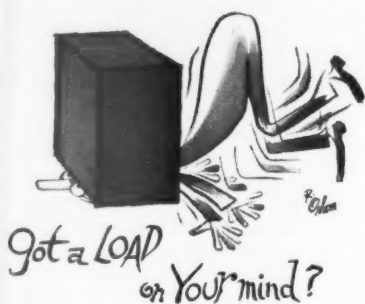
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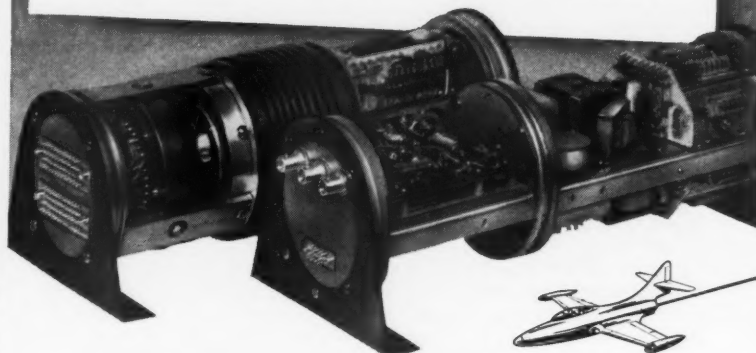
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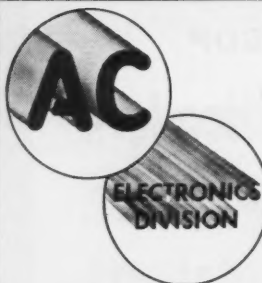
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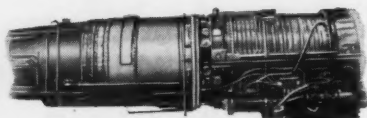
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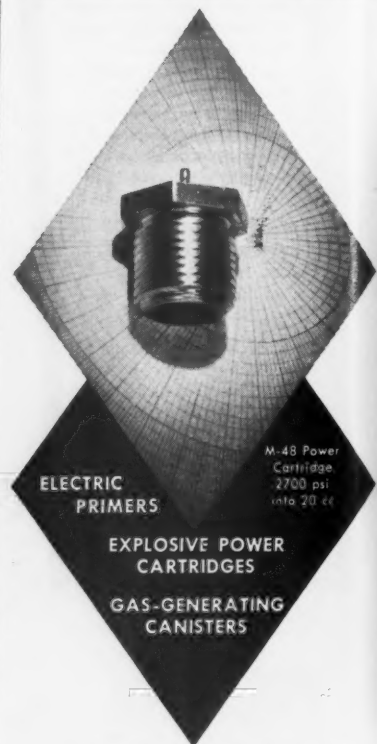
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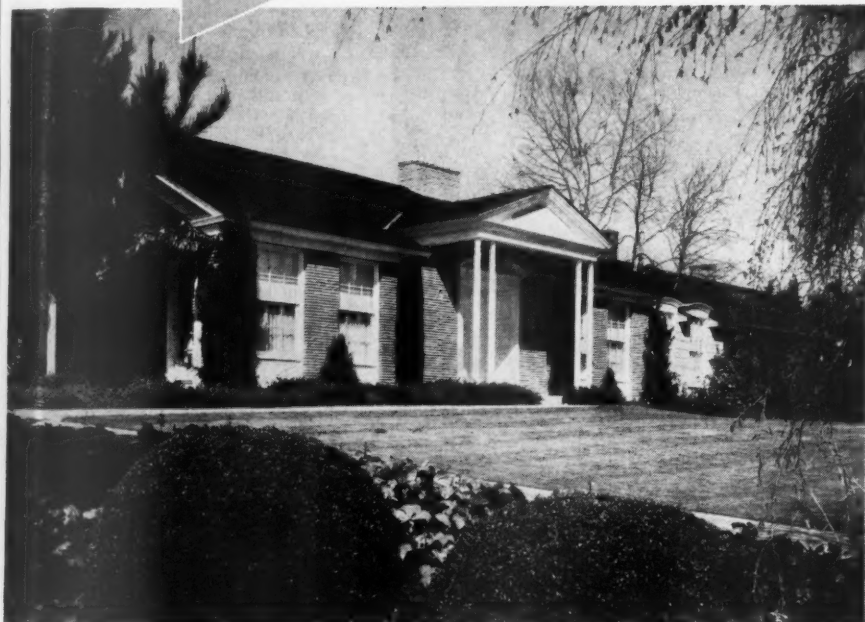
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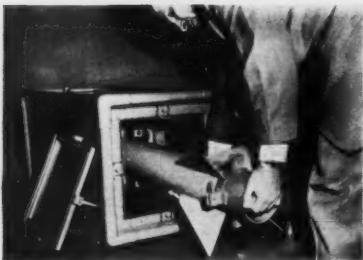
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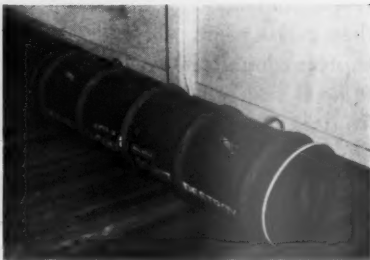
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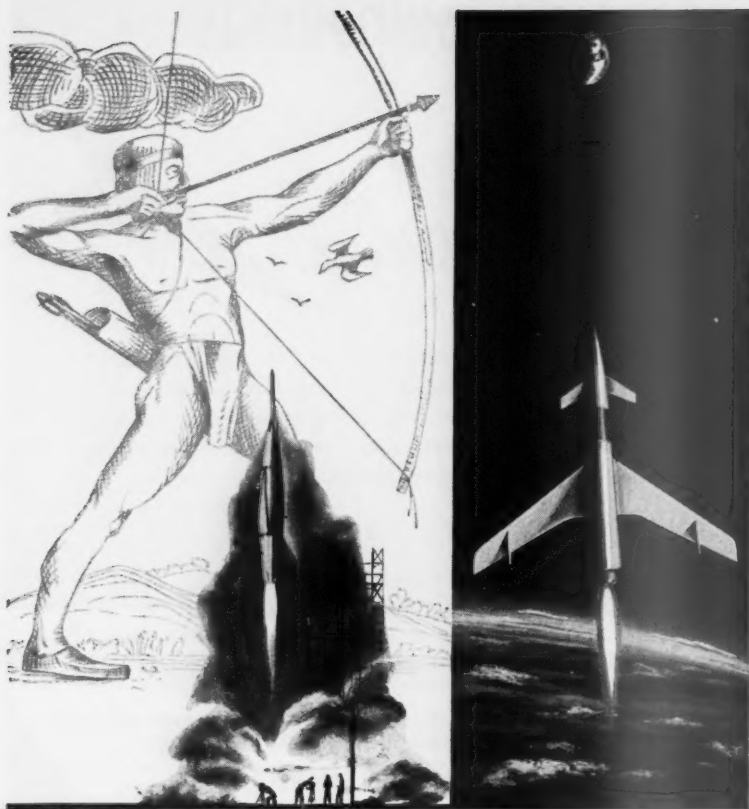
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Jet Propulsion Engines

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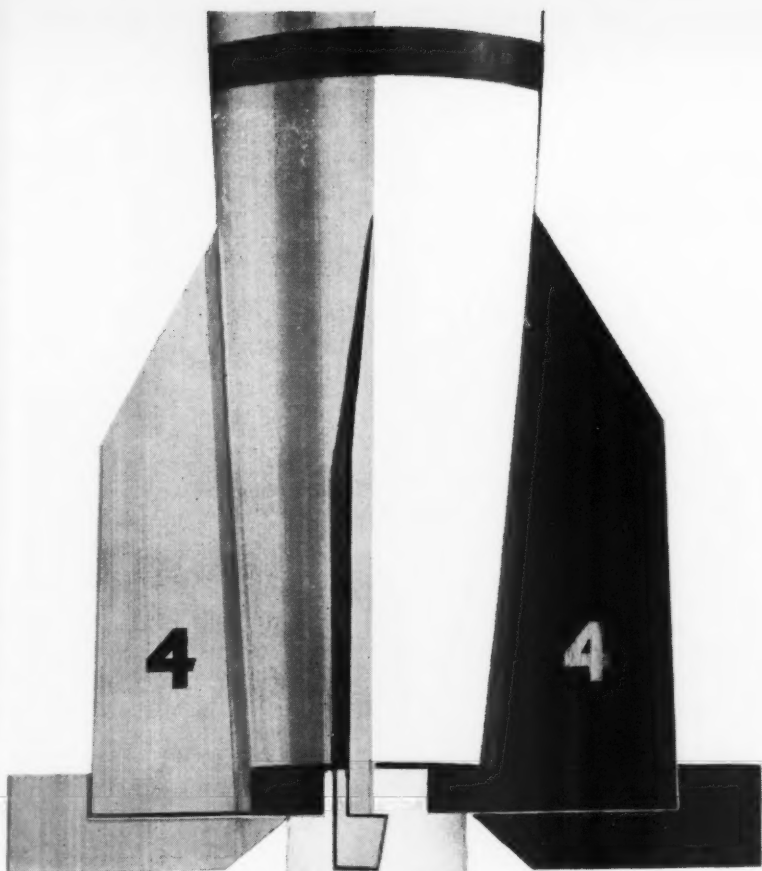
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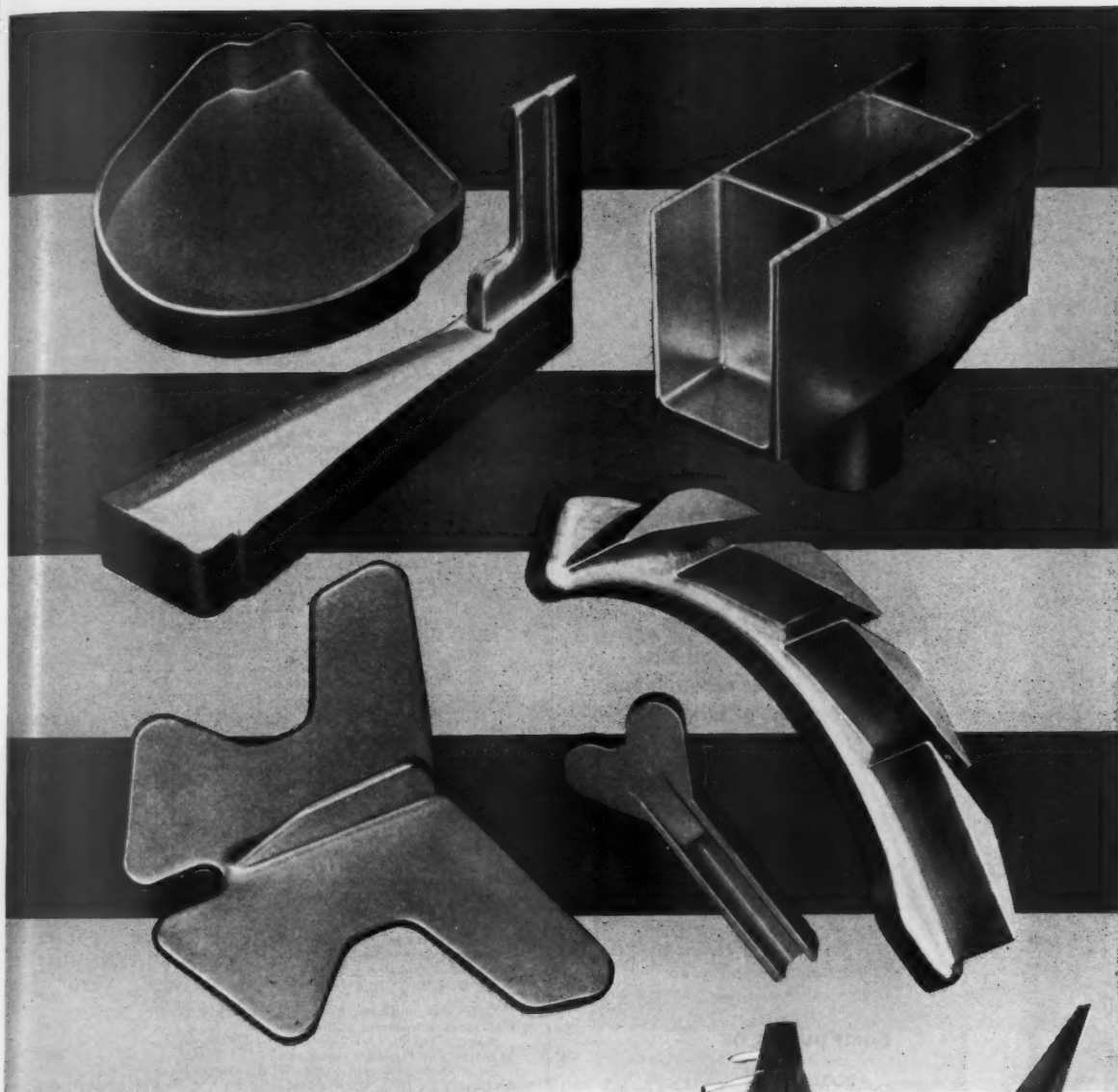
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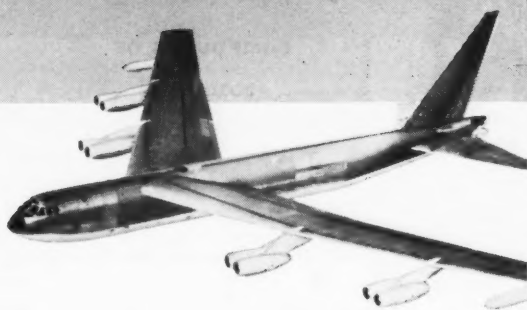
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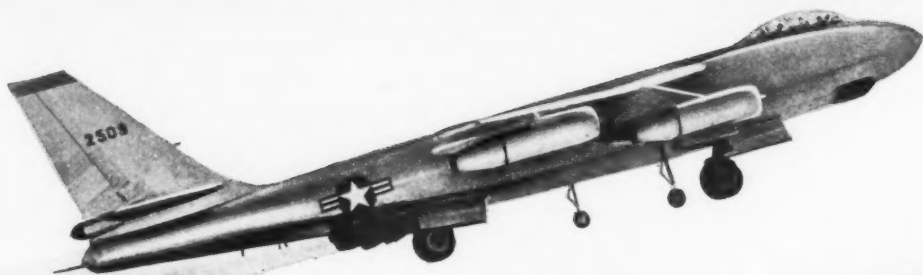
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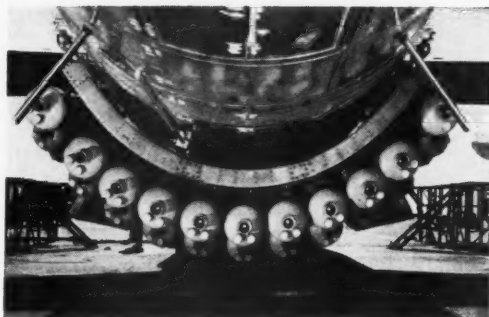
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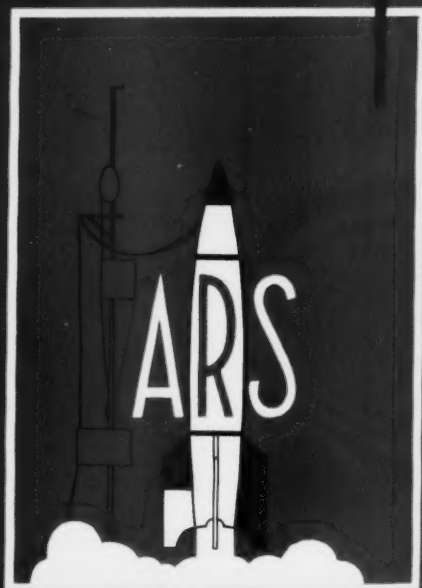
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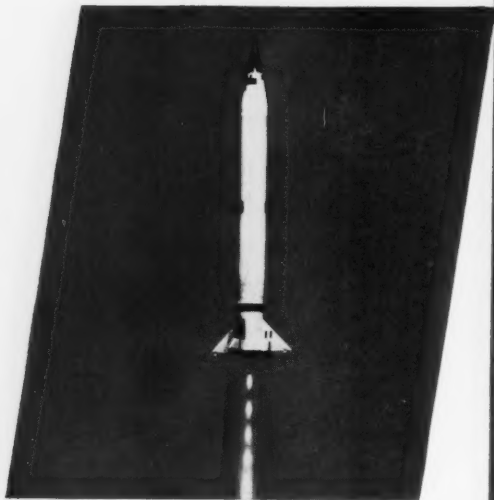
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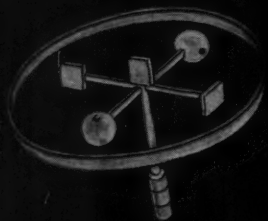
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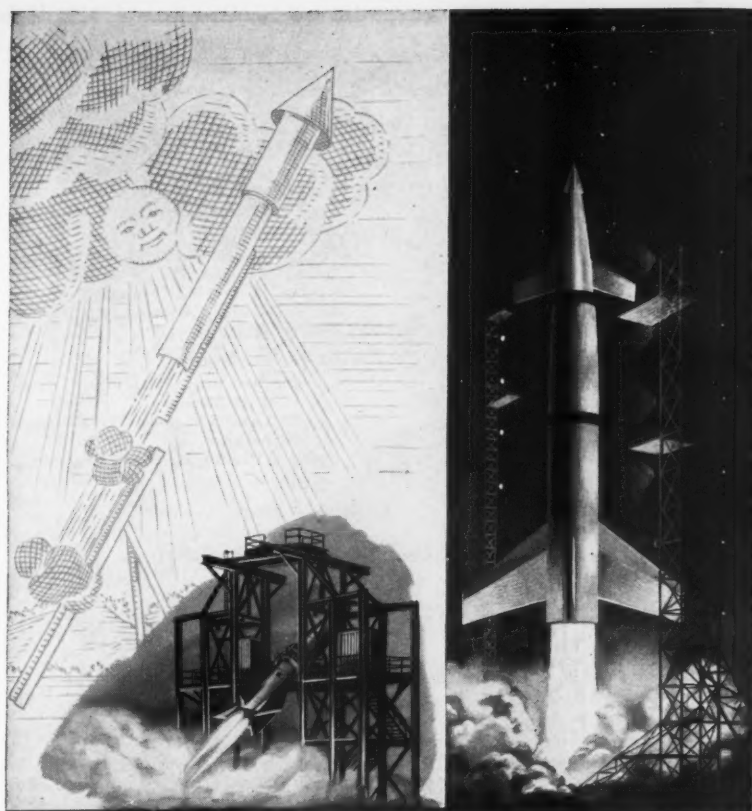
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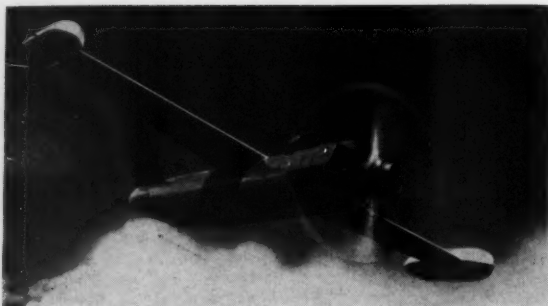
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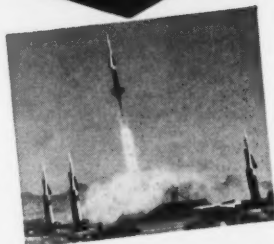
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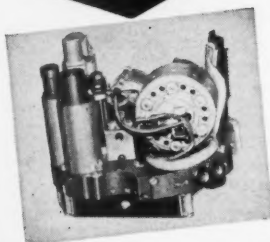
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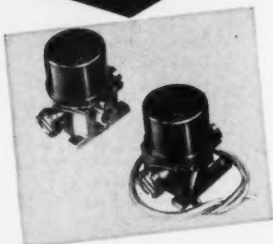
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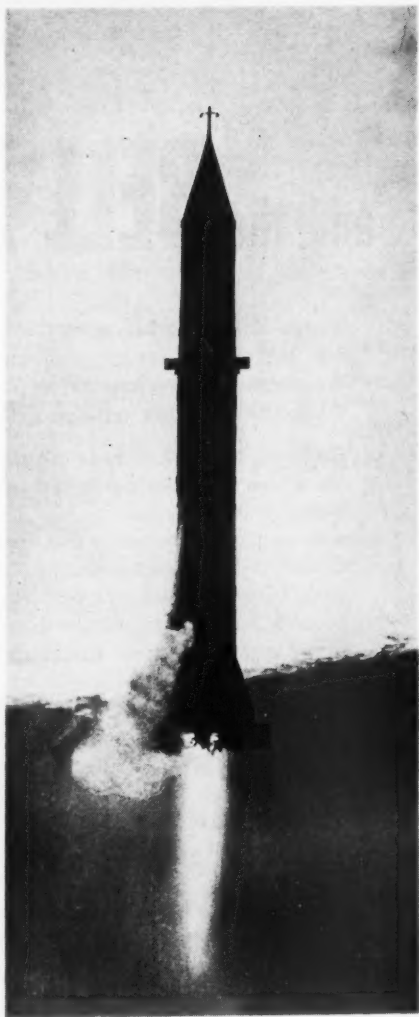
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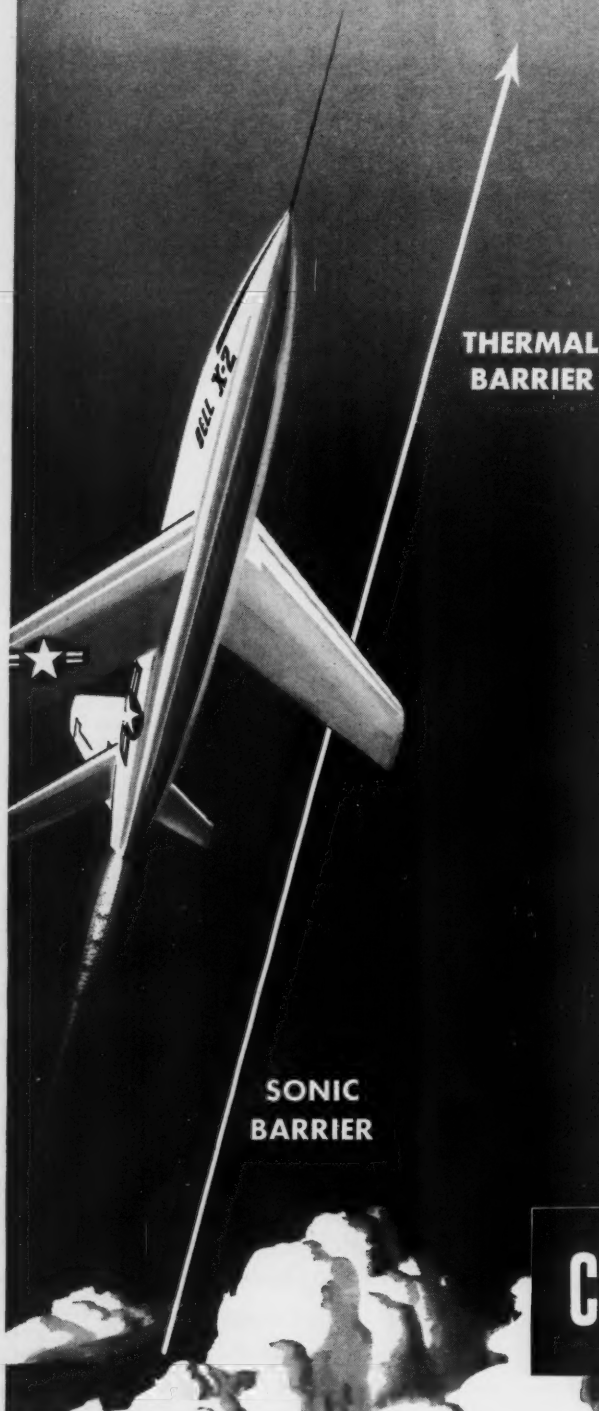
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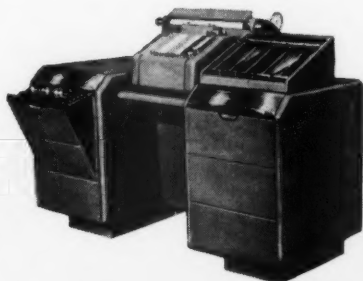
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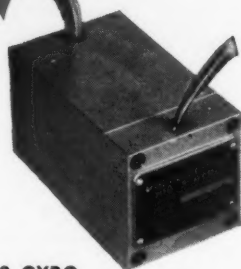
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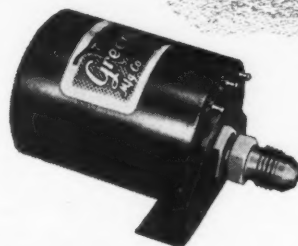
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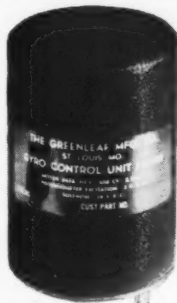
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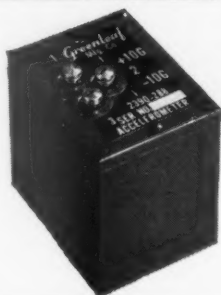
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Helping Make Naval History

Terrier-Armed Cruiser to Defend Navy's Atlantic Fleet Task Forces

BY DAVID A. ANDERTON

Aviation Week, November 7, 1955.—Operational integration of the Navy's first guided missile cruiser—the USS *Boston* (CAG-1)—will begin early next year when the converted ship joins the Atlantic Fleet.

The *Boston* is armed with two twin launchers firing the supersonic Convair Terrier anti-aircraft missile: Fleet operations with the Terrier follow extensive test firings of the missile, including a series from the Atlantic Fleet's experimental gunnery ship, the USS *Mississippi*.

TERRIER HISTORY

The Terrier is the first weapons system to come from the Bureau of Ordnance's ten-year-old Section "T" contract with the Applied Physics Laboratory of Johns Hopkins University. Prompted by Japanese Kamikaze attacks in the closing months of World War II, the bureau established the missile research and development project under the code name of Bumblebee.

Scientists of JHU/APL made initial studies that led to the Terrier; other phases of the Bumblebee program have produced Talos, another anti-aircraft missile with ramjet propulsion, and the Triton.

As Terrier tests made the missile look more and more promising, the Navy opened a new production plant at Pomona, Calif., and turned the operation of the facility over to the Convair Division of General Dynamics. Production began in January 1953.

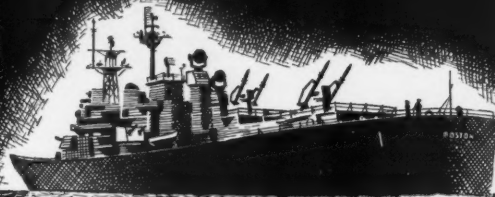
MISSILE DIVISION

The powerplant, a sustainer rocket built by the M. W. Kellogg Co., occupies the bay between the wings and the tail. The aft guidance section mounts the fins, antennas and related avionic gear.

Convair produces both guidance sections, the spines connecting those sections, and all the aerodynamic surfaces.

The *Boston*, as the first guided-missile cruiser, forms the nucleus of the Navy's first guided-missile division.

In addition to actually building the Terrier's powerplant, as mentioned in this "Aviation Week" article, The M. W. Kellogg Company also participated in the initial conception of the Terrier's propulsion system and in the ultimate development of this guided missile as one of the Navy's primary anti-aircraft weapons. To date, M. W. Kellogg's unique industrial background in the utilization and control of high temperatures and pressures has contributed to the engineering and production of over ten different rocket engines for the Armed Forces.



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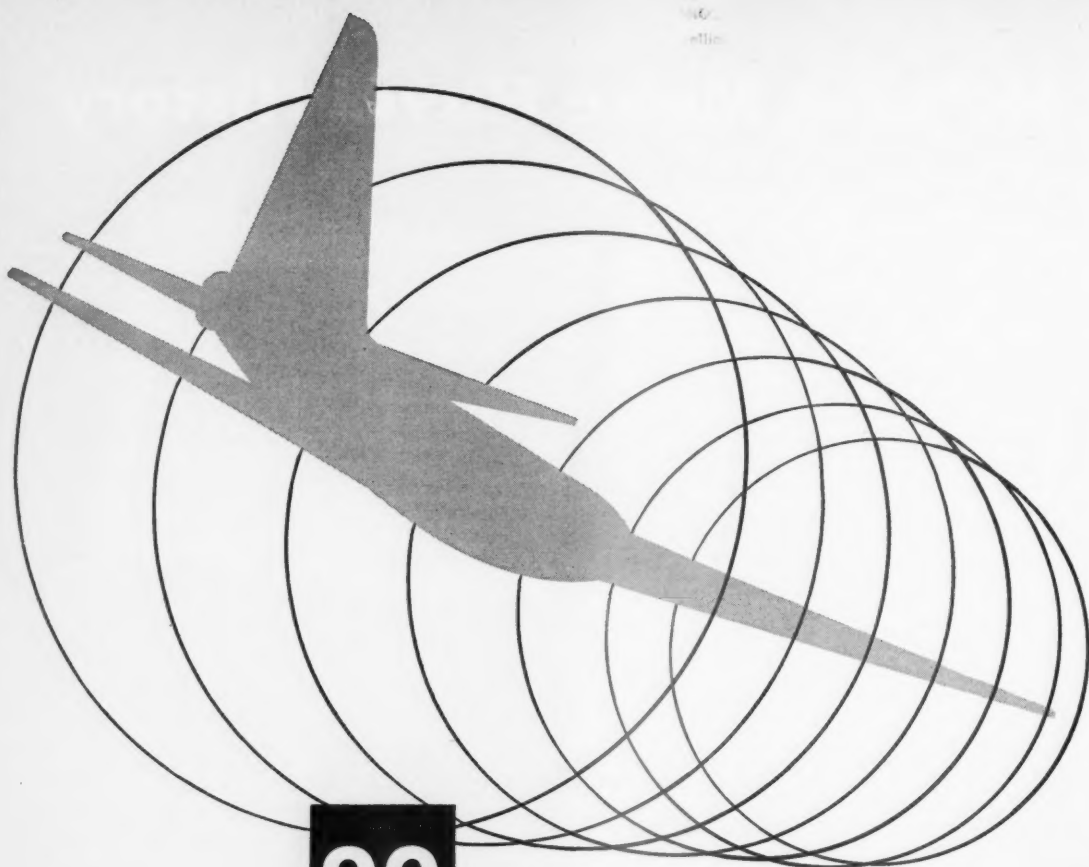
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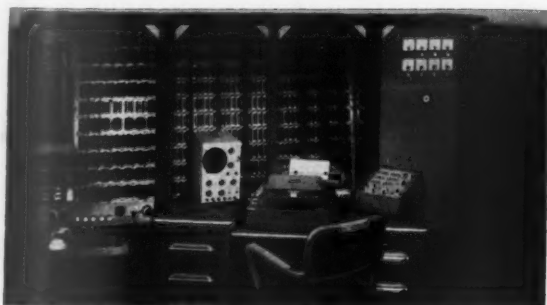
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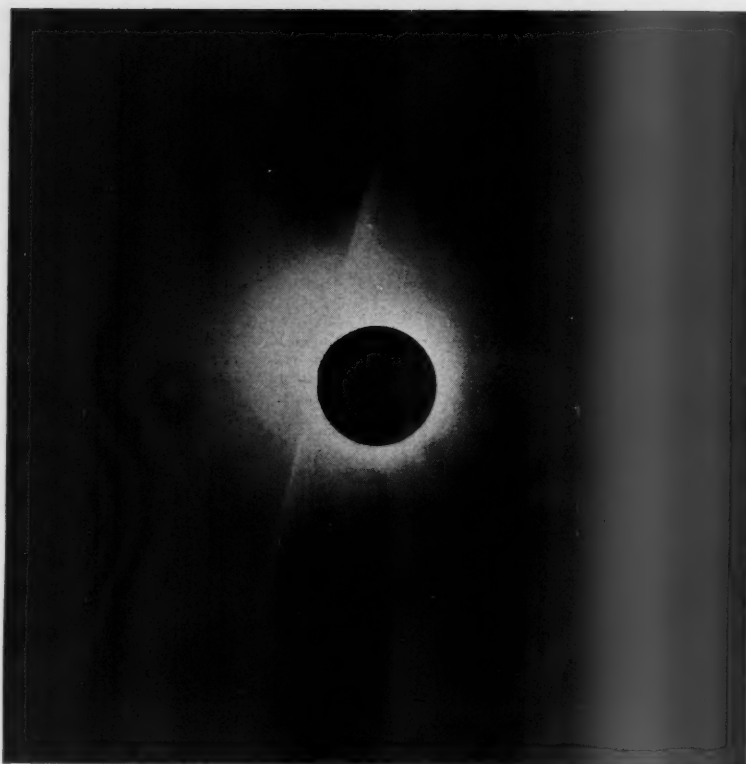
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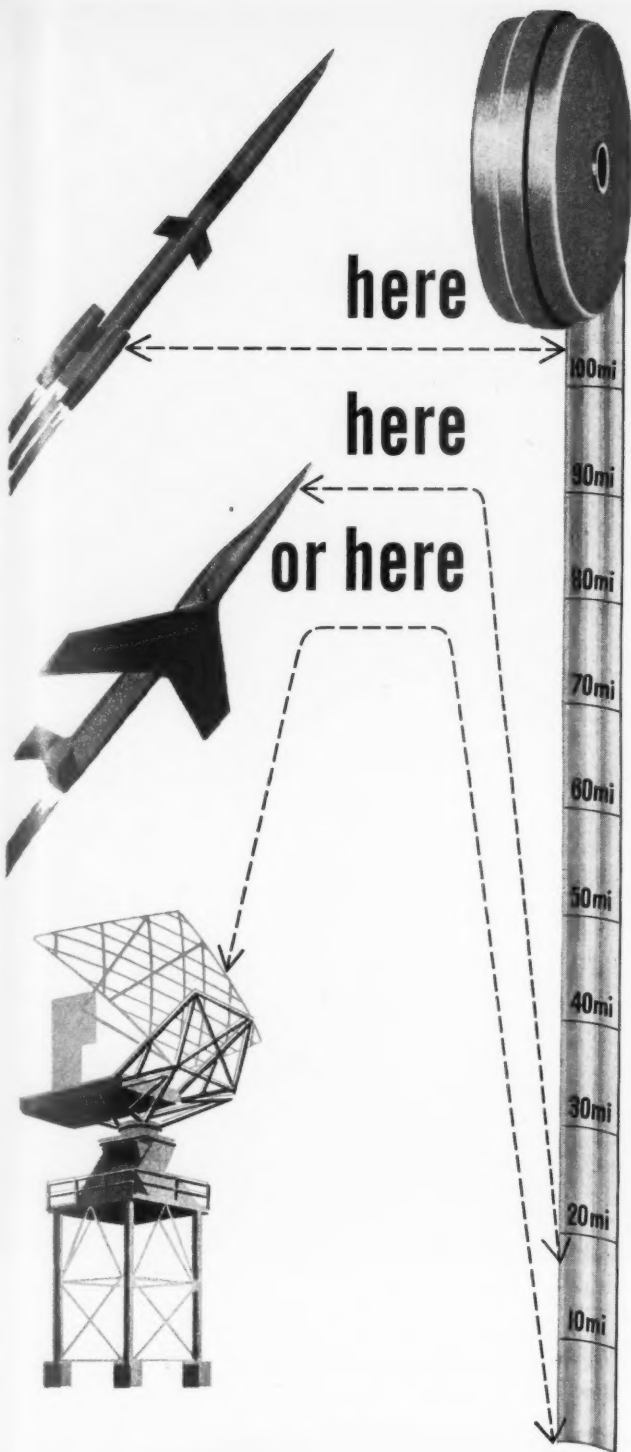
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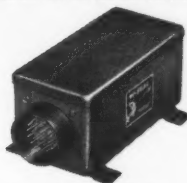
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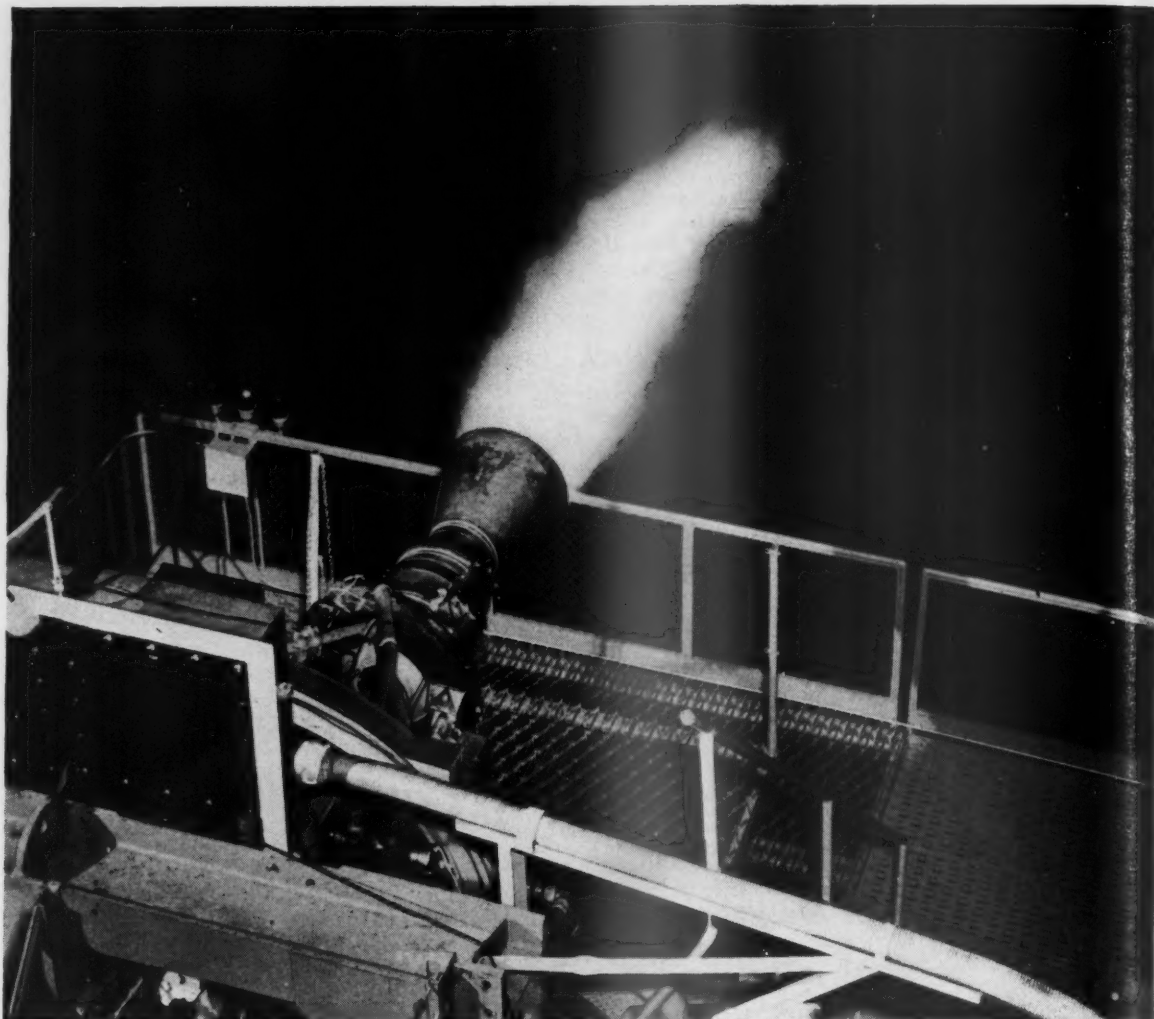
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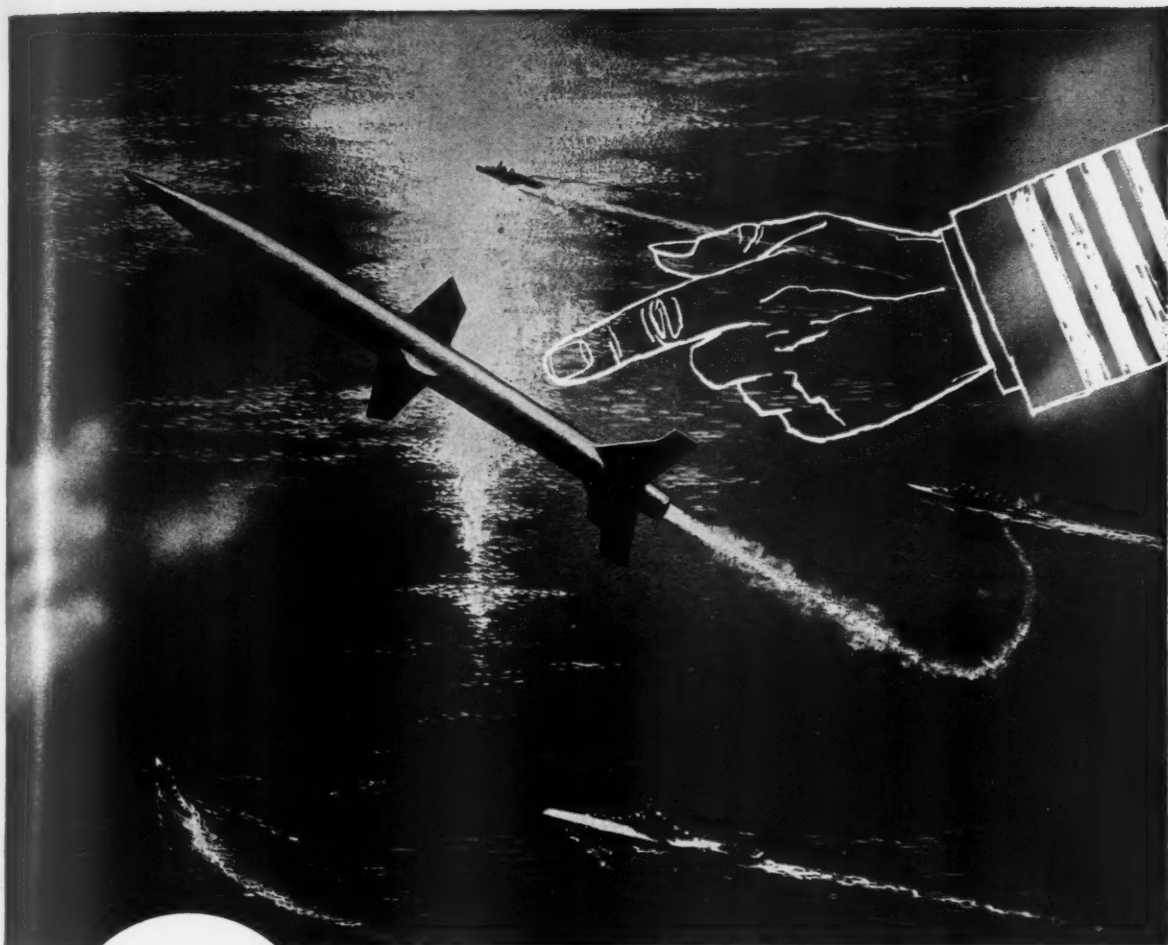
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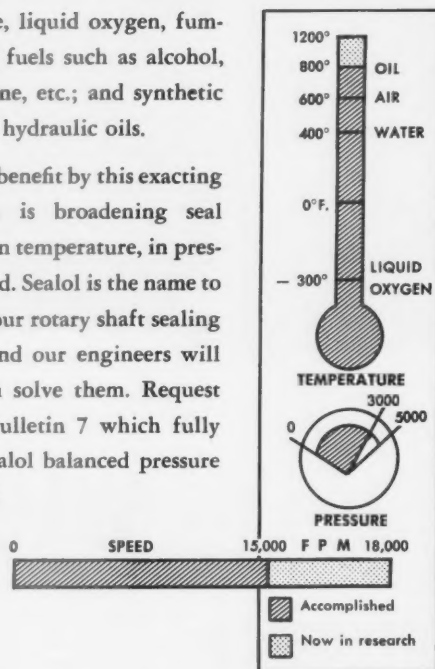
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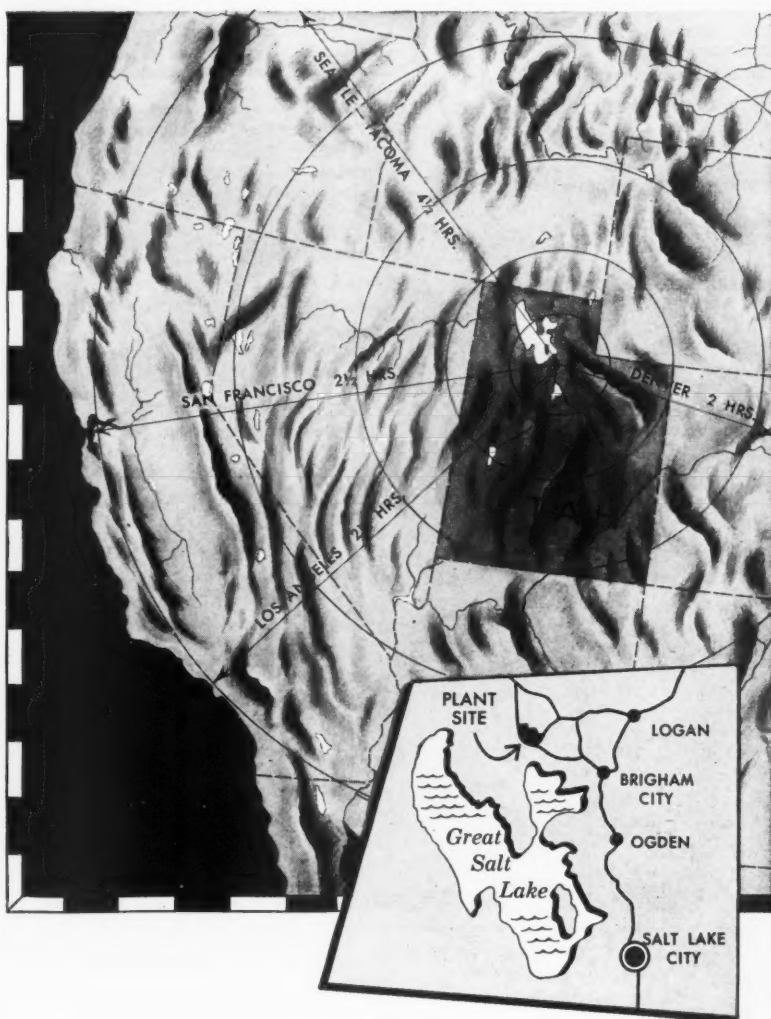
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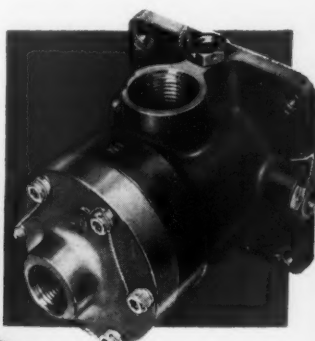
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Haurian, Peter N., Chief, Military & Industrial Services
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Lang, Theodore E., Draftsman-Designer
Nielsen, Julian J., Engineer
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Peebles, Lester H., Development Engineer
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Richardson, Al, Technical Writer
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Wilson, Earl L., Development Engineer
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 Tech. Res. Inst. of Nat'l Safety Agency, Tokyo
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 Usis Presentation, American Embassy, Tokyo

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Liberia Mining Co., Ltd., Monrovia

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Gravenhage
Koninklijke Shell Laboratorium, Delft
Ministry of War, Bureau Documentation, Hague
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AMERICAN ROCKET SOCIETY

500 Fifth Ave., New York 36, N. Y.

A national association for the advancement
of rocketry, jet propulsion, and astronautics

Aims and Functions

THE AMERICAN ROCKET SOCIETY is a national association of engineers and scientists devoted to "the development and application of the principle of jet propulsion as applied to rockets, aircraft, water and underwater craft and to all other appropriate and practical devices" and to "the development of the sciences and engineering techniques pertaining thereto" such as instrumentation, guidance and control, high temperature materials, upper atmospheric research, aerodynamics, structures, combustion, fuels and propellants, heat transfer, etc. ARS has always maintained a deep interest in responsible scientific study of space flight and in fostering sound professional and public interest in this subject. The Society carries out its aims through:

1. Section Meetings, held periodically at most of the centers of rocket and jet propulsion activity throughout the country. Sections are self-governing and programs are both technical and social.

2. Regional Meetings, which include technical sessions as well as field trips to rocket, guided missile and aircraft establishments, observatories, etc. These meetings are often held in cooperation with other technical societies.

3. The Annual ARS Convention. Held in conjunction with the ASME Convention, this gathering includes technical sessions during which original papers are presented on all aspects of the Society's interests. Outstanding guest speakers are presented at luncheon and dinner meetings and the ARS Awards are presented. Awards include the Robert H. Goddard Memorial Award for work in liquid propellants, the C. N. Hickman Award for solid propellants, the James H. Wyld Memorial Award for outstanding application of rocket power, the ARS Astronautics Award for contribution to the advance toward space flight, the G. Edward Pendray Award for rocket and jet propulsion literary effort, and the ARS Student Award for the best student paper on rocket and jet propulsion. Fellowships are also presented at this meeting to leading figures in the field.

The ARS Library, located in New York, contains historical as well as current material relating to rocket progress.

An affiliation exists between the American Rocket Society and the American Society of Mechanical Engineers which permits ARS members to use the Engineering Library in New York, attend ASME Sectional and National meetings and allows the purchase of reprints of articles at member rates.

History and Accomplishments

ARS was organized on March 21, 1930. During its early years, experimental rocket firings were made, and from them evolved data and equipment which led to the development of several significant rocket engines, as well as to the formation of industrial enterprises which today hold distinguished places in the field.

As progress in rocket and jet propulsion became more rapid, and demands from the military mounted, the Society devoted itself exclusively to dissemination of information through meetings and publications.

Dr. Robert H. Goddard was one of the early members of

the society, as were many other pioneers who now hold key positions in industry, science, education and government.

Membership

Four types of membership are open in the society:

MEMBER: Members shall consist of engineers and scientists who are actively engaged in the development or application of rocket or jet propulsion, other persons who have been working on the development or application of rocket or jet propulsion for at least four years and who hold or have held responsible positions in these fields, and such persons as may be deemed eligible for this class of membership by the Board of Directors by virtue of their outstanding accomplishments in other fields and their unusual interest in the purposes of the Society.

ASSOCIATE MEMBER: Associate members shall be persons, other than students, who are actively interested in the development or application of rocket or jet propulsion.

STUDENT MEMBER: Student members shall be persons not less than 17 years of age whose principal occupation is study at a recognized educational institution or who are serving as enlisted personnel in the Armed Forces of the United States, and who are interested in the development or application of rocket or jet propulsion.

CORPORATE MEMBER: Corporate Members shall be educational, scientific or industrial organizations who may choose this method of expressing their interest in the development or application of rocket or jet propulsion, and who are considered acceptable by the Board of Directors. Each Corporate Member shall be entitled to five representatives with the rights and privileges of Members.

Subscriptions to the Journal without membership are available at the rate of \$12.50 per year.

Certificates of membership are sent to each member upon acceptance. Identification pins may be purchased for \$2.00 from the office of the Secretary.

Publications

JET PROPULSION, Journal of the American Rocket Society, is published monthly and mailed to all members. Edited by distinguished scientists, the Journal is recognized as the most complete and authoritative source of original technical material on rocketry, jet propulsion, astronautics, and allied sciences.

It also includes feature articles on industry, government, and university activities, news items, and ARS happenings. It carries a calendar of coming meetings, a digest of books and articles published on rocket and jet propulsion in this country and abroad, and reviews by authorities on current books.

Information on advertising rates for the Journal can be obtained from the Secretary.

BOOKS AND TECHNICAL PAPERS. The Society makes available to its members at special rates, technical papers, reprints and other useful information bearing on jet propulsion, rockets and their application. It also has available back issues of the Journal and its predecessor, *Astronautics*.

A list of such material currently available may be obtained by writing to the office of the Secretary

PLEASE USE FORM ON REVERSE SIDE TO APPLY FOR MEMBERSHIP

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I hereby apply for membership in the American Rocket Society, Inc. and enclose \$.....
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